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LEARJET 60
PILOT TRAINING MANUAL

THIRD EDITION

REVISION 0.1

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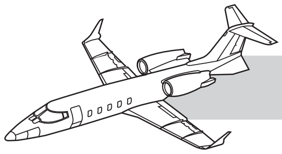

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AIRCRAFT GENERAL

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CHAPTER 1
AIRCRAFT GENERAL

INTRODUCTION

This training manual provides a description of the major airframe and engine systems installed in the Learjet.

The material in this manual applies to the Learjet 60 model.

The *Pilot Training Manual Volume 2* is based on information from the documents listed in Table 1-1.

Table 1-1. ASSOCIATED DOCUMENTS

TITLE	IDENTIFIER
Learjet 60 Airplane Flight Manual	FM-123
Learjet 60 Crew Checklist and Quick Reference Handbook	CL-123C
Learjet 60 Pilot Manual	PM-123
Master Minimum Equipment List	

This chapter covers the structural makeup of the aircraft and gives a general description of the systems. No material is meant to supersede or supplement any of the manufacturer's system or operating manuals.

The material presented has been prepared from the basic design data, and all subsequent changes in aircraft appearance or system operation will be covered during academic training and in subsequent revisions to this manual.

GENERAL

The Learjet 60 (Figure 1-1) is certificated under 14 CFR Part 25 as a transport category aircraft designed for all-weather operation at all altitudes up to 51,000 feet.

STRUCTURES

GENERAL

The Learjet 60 airframe was designed with great consideration for safety. All load-bearing components can withstand 90% of design G-loading with the failure of the adjacent structural component. The structure consists of the fuselage, the wing, the empennage, and flight controls. The discussion on the fuselage includes all doors and windows. Figure 1-2 shows the general dimensions of the basic aircraft.

Figure 1-3 displays the Lear 60 turning radius, based upon the maximum pilot input available, at a nosewheel orientation of 60° from center at slow speed (2 knots or less).

Figure 1-4 is the manufacturer's display of danger areas around the Learjet 60. Areas portrayed represent the weather radar transmission cone as well as sections in front of the engine and aft of the engine exhaust cone.

FUSELAGE

General

The fuselage is constructed of all-metal-clad stressed skin with stringers that provide higher structural integrity at a lighter weight. It employs the "area rule" design to reduce aerodynamic drag, and has four basic sections (Figure 1-5). They are:

1. The nose section, which extends from the radome aft to the forward pressure bulkhead.

2. The pressurized section, which includes the cockpit and passenger areas, extends aft to the rear pressure bulkhead.

3. The fuselage fuel section starts just aft of the rear pressure bulkhead and extends to the tailcone.

4. The tailcone section contains the portion of the aircraft aft of the fuel section.

The fuselage also incorporates attachments for the wings, tail group, engine support pylons, and the nose landing gear.

In addition to the cockpit and passenger compartments, the fuselage includes a nose wheel well, unpressurized aft (external) baggage compartment, and an unpressurized tailcone equipment bay.

Figure 1-1. Learjet 60

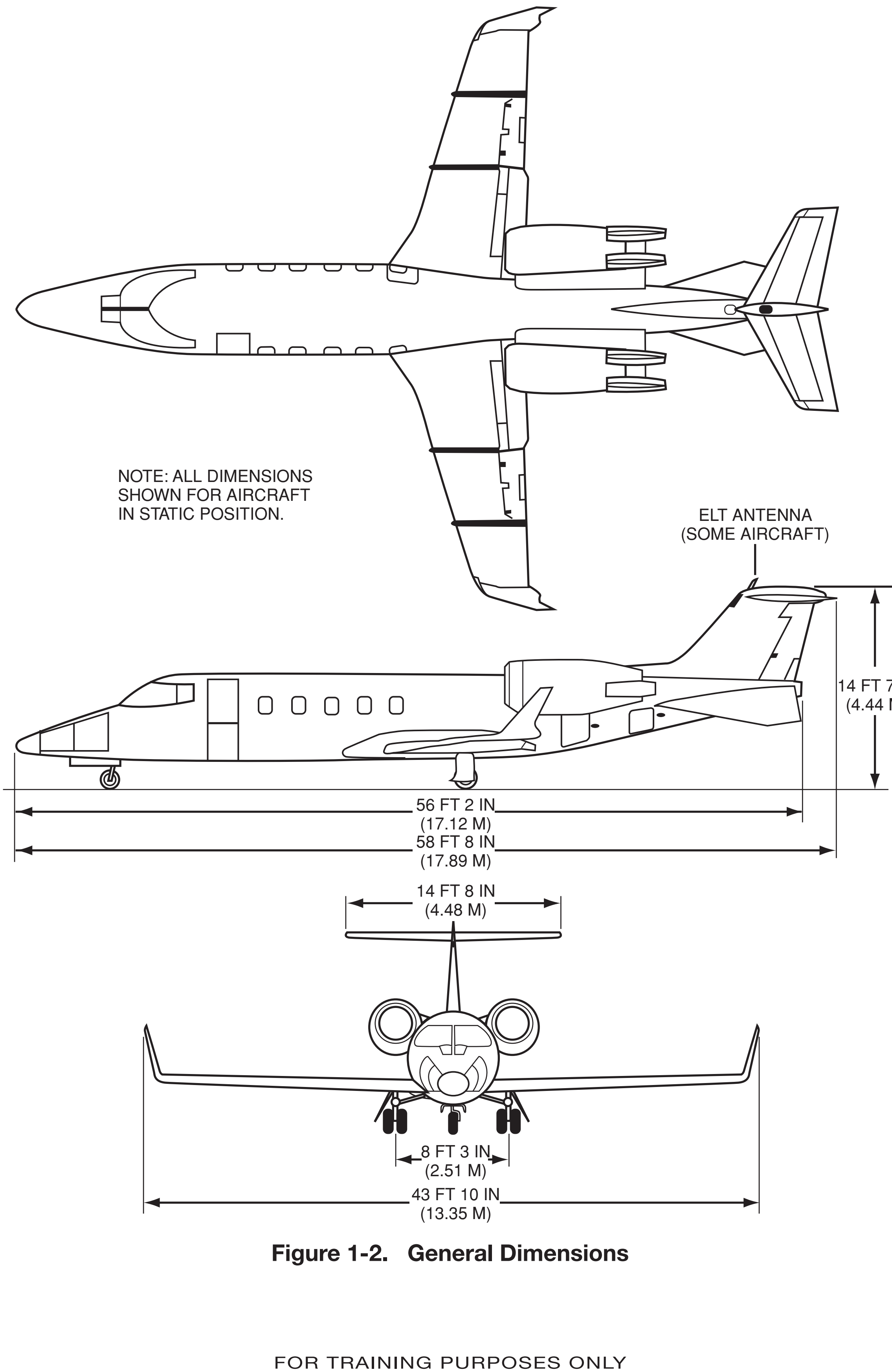
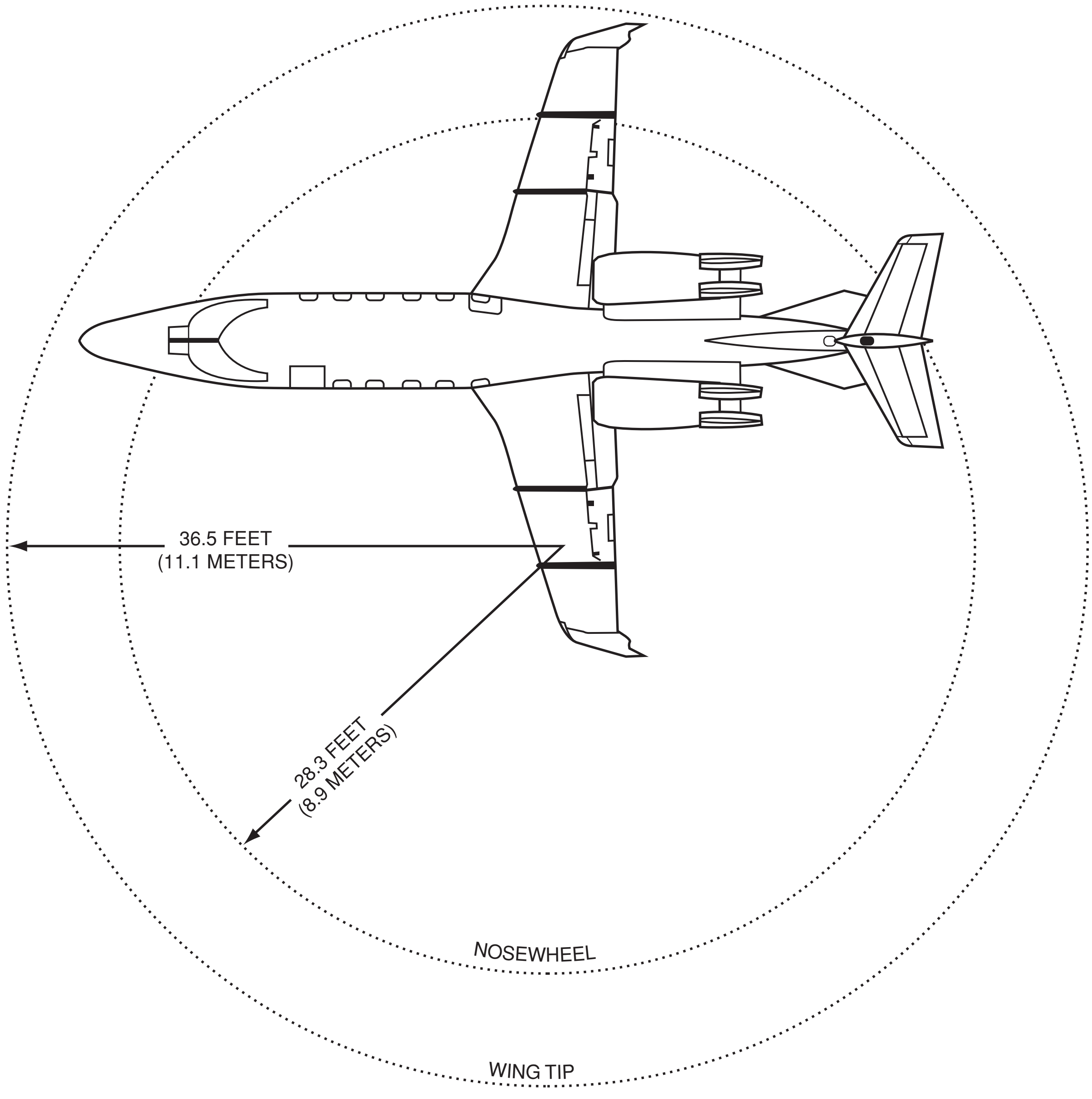


Figure 1-2. General Dimensions



NOTE:

TURNING RADIUS EXPRESSED ABOVE IS BASED UPON 60° NOSEWHEEL TRAVEL (FULL-AUTHORITY/LOW-SPEED STEERING). LIMITED AUTHORITY STEERING PROVIDES 24° OF NOSEWHEEL TRAVEL. TURNING RADIUS WILL INCREASE ACCORDINGLY.

Figure 1-3. Turning Radius

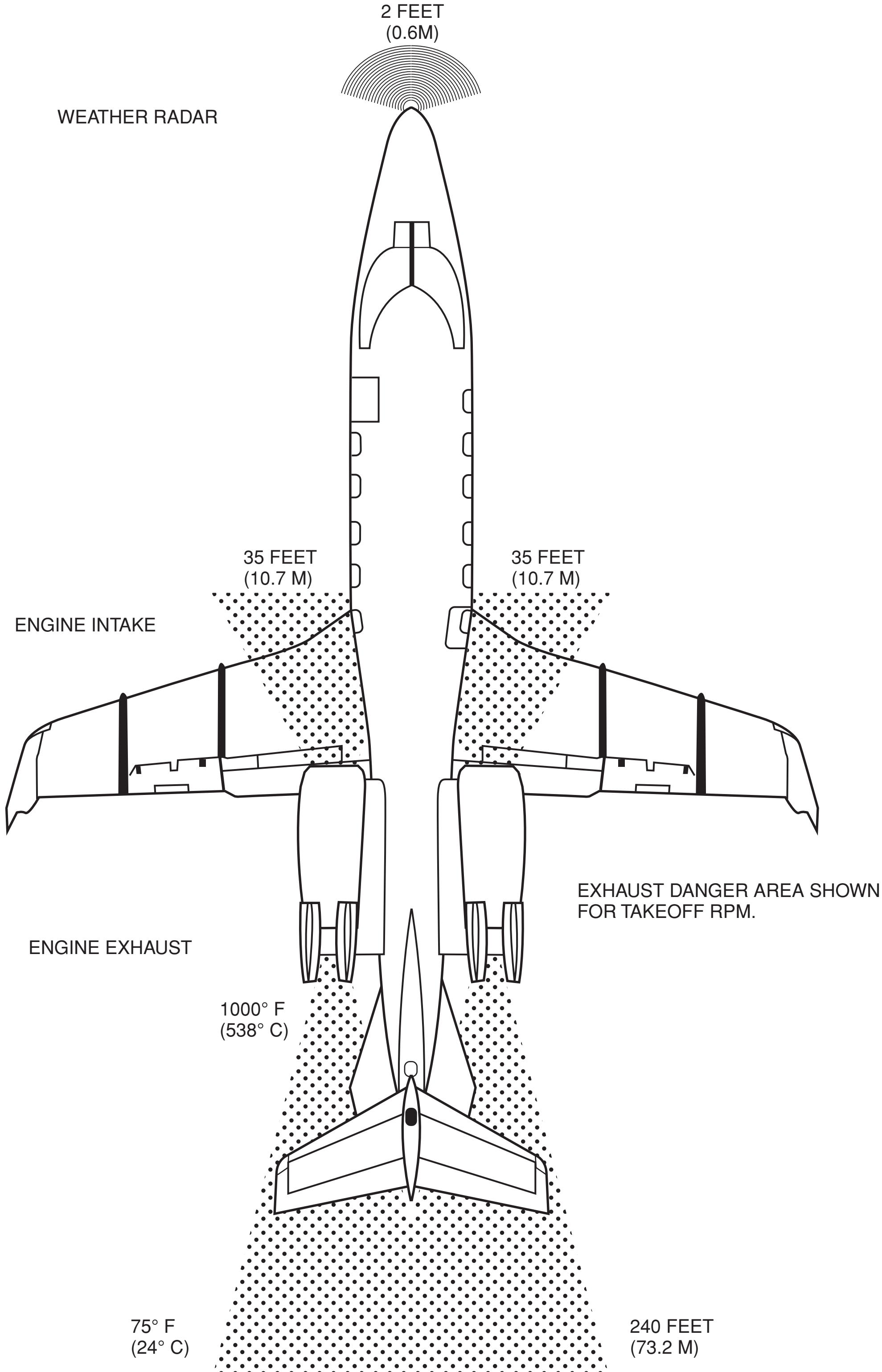


Figure 1-4. Danger Areas

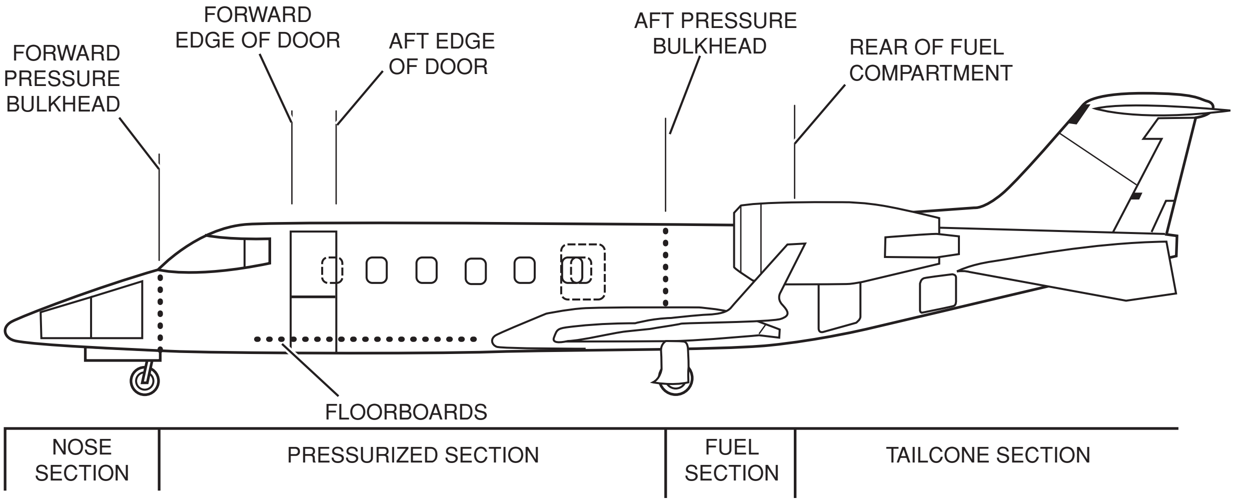


Figure 1-5. Fuselage Sections

Nose Section

The nose of the fuselage (Figure 1-6 and Figure 1-7) is the radome. Aft of the radome there are two access doors on each side of the nose. The doors provide access to the avionics equipment bay.

The access doors are fastened closed with CAMLOCS.

Figure 1-6. Nose Section—Avionics Access Panels (Left Side)

Pressurized Section

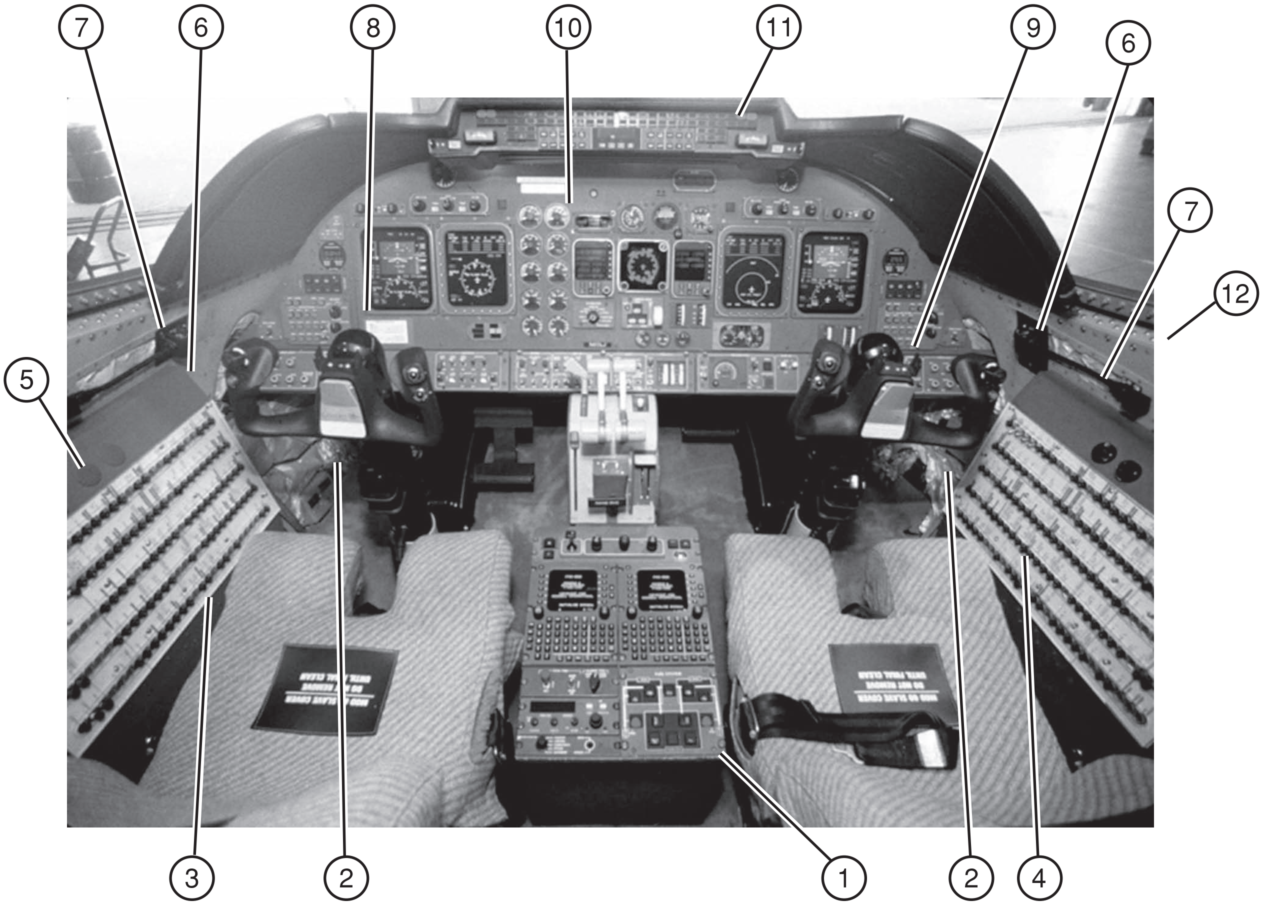
The pressurized area is between the forward pressure bulkhead and the aft pressure bulkhead. This section includes the cockpit and the passenger area and includes a lavatory. A baggage area is at the aft end of the cabin. The cockpit (typical) is indicated on Figure 1-8.

The instrument panel (typical) is as indicated on Figure 1-9.

The passenger/crew door is located on the left side of the fuselage just aft of the cockpit. An emergency exit/baggage door is on the right side, over the wing leading edge. There are 11 cabin windows: five on the left side, and six on the right. The aft right window is in the emergency exit/baggage door.

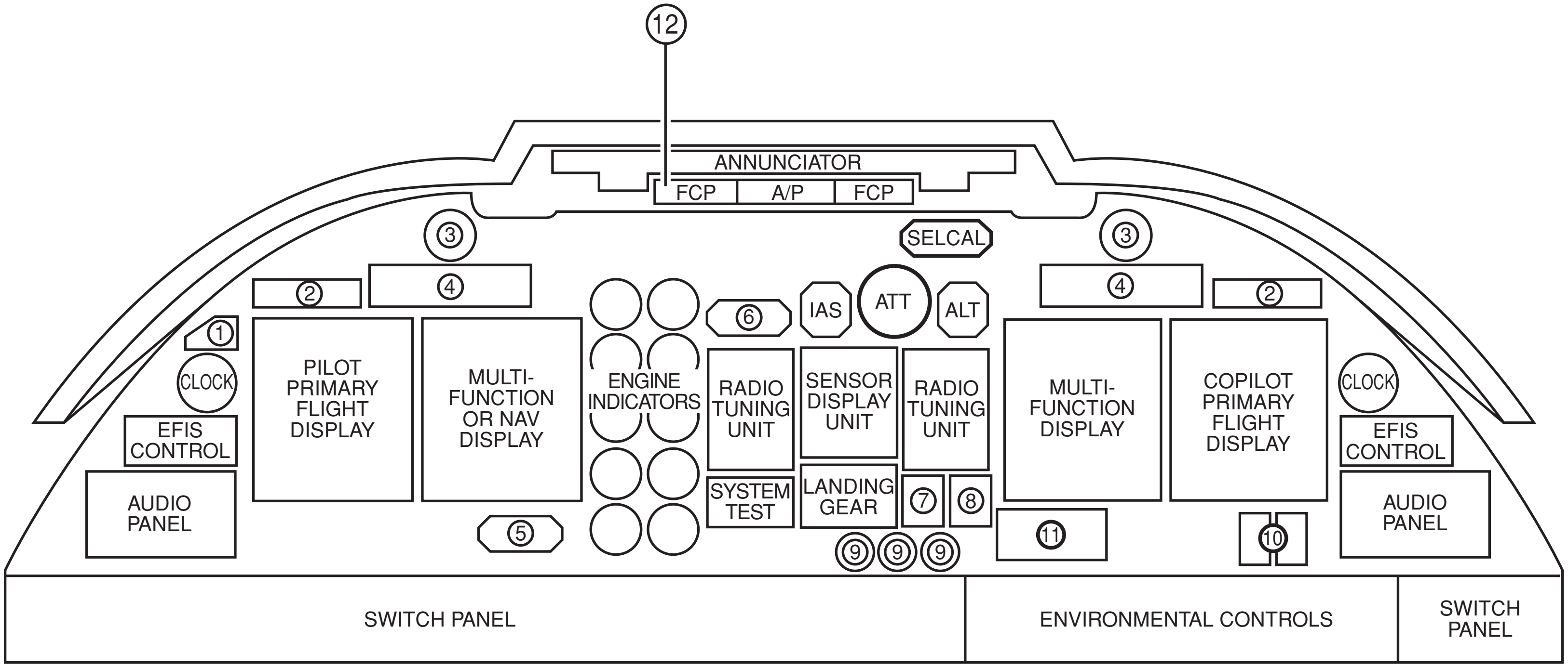
The cockpit (Figure 1-8) seats two pilots, side by side. There is a large, curved windshield in front of each pilot position, and a side window just aft. The pilot side window may be opened on the ground (some aircraft).

Figure 1-7. Nose Section—Avionics Access Panels (Right Side)



1. CENTER PEDESTAL AND THROTTLE QUADRANT
2. AIR OUTLET (ANKLE)
3. PILOT'S CIRCUIT BREAKER PANEL
4. COPILOT'S CIRCUIT BREAKER PANEL
5. OXYGEN CONTROLS AND MIC/PHONE JACK PANEL
6. FOLDOUT WORK TABLE
7. MAP LIGHT
8. PILOT'S CONTROL COLUMN
9. COPILOT'S CONTROL COLUMN
10. INSTRUMENT PANEL
11. ANNUNCIATOR PANEL
12. COPILOT'S MIC/PHONE JACK PANEL

Figure 1-8. Cockpit Layout



1. ANTI-SKID AND PARKING BRAKE ANNUNCIATOR LIGHT
2. ALTITUDE AWARENESS PANEL (AAP)
3. ANGLE OF ATTACK INDICATOR (AOA)
4. AIR DATA REFERENCE PANEL (ADRP)
5. ELECTRICAL POWER MONITOR (EPM)
6. FUEL QUANTITY INDICATOR
7. SPOILER POSITION INDICATOR
8. FLAPS POSITION INDICATOR
9. TRIM INDICATORS
10. CABIN/COCKPIT TEMPERATURE INDICATORS
11. CABIN PRESSURIZATION INDICATORS
12. AUTOPILOT AND FLIGHT DIRECTOR CONTROL PANELS

Figure 1-9. Instrument Panel (Typical)

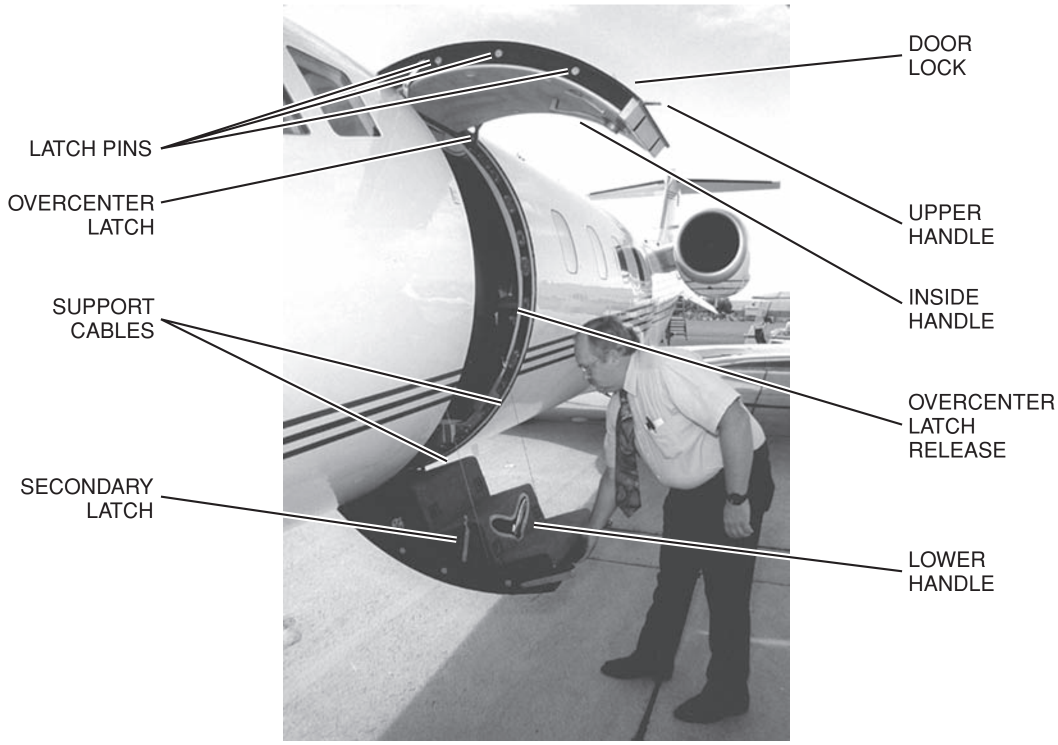


Figure 1-10. Cabin Entry Door (Open)

Doors

Cabin Entry Door

The cabin entry door is the primary means of access and egress for the passengers and the crew (Figure 1-10). The 25-inch wide door has a clamshell design, each half hinged to the fuselage. The upper half serves also as an emergency exit. The lower half has integral entrance steps.

The upper portion of the door has both outside and inside locking handles which are fastened to a common shaft through the door.

Rotating either of these handles to closed drives six locking pins into the fuselage frame (three pins forward and three aft) and two pins through interlocking arms that secure the door halves together (Figure 1-11).

The lower door has a single locking handle on the inside of the door.

Figure 1-11. Cabin Entry Door (Closed)

Rotating the lower door handle to the closed (forward) position drives four pins into the fuselage frame (two forward and two aft). There are a total of 12 locking pins on the 2 door sections. When the door handles are in the closed position, the pins make contact with microswitches. If any of the switches are not tripped when the door handles are closed, a red ENTRY DOOR light on the main annunciator panel will flash. If the lower handle and the upper handle are both in the open position, the ENTRY DOOR light will be on steady (Figure 1-12).

Figure 1-12. Door Warning Lights

If the ENTRY DOOR light is flashing while the door is closed, the pilot can visually check through inspection ports (Figure 1-13) for proper alignment between the white lines on the latch pins and on the door structure. Four pins in the lower door may be viewed by lifting carpet tabs to reveal the inspection ports.

The two latch pins that connect the upper and lower doors are visible through the upholstery gap at the interface and do not have white lines.

Figure 1-13. Door Latch Inspection Ports and Inside Locking Handle

The upper door has a torsion bar to provide opening assistance. On aircraft SN 60-067 and subsequent, two gas struts replace the torsion bar. A latch, when overcentered, retains the door in the open position. The overcenter latch can be released for door closing with a door latch release handle just inside the door on the lower aft side of the door frame (Figure 1-10).

A keylock provides positive outside security for the upper door locking mechanism, but must not be employed when the aircraft is occupied since the upper door is an emergency exit. To alert the crew to this situation, the red ENTRY DOOR light will flash when the keylock is engaged.

The lower door is equipped with two cables connected to take-up reels in the door, which assist in pulling the door closed and also prevent damage if the door has a sudden opening drop. The latter is also assisted by a self-contained hydraulic damper in the lower door. A T-handle/cable allows the lower door to be closed from inside the aircraft.

A secondary safety latch is installed on the lower door and is separate from the door locking system. It consists of a notched pawl attached to the door, and it engages a notched striker plate attached to the aircraft frame when the door is closed. This engagement holds the lower door closed while the locking handle is being positioned to the locked position; it also keeps the door from falling open as soon as the door handle is positioned to open. The latch is released from the inside by lifting the pawl or from the outside by depressing a release button flush-mounted in the lower door skin.

A rubber seal fastened around the doorframe has holes which allow the entry of pressurized cabin air, forming a positive seal around the door.

Emergency Exit/Baggage Door

The emergency exit/baggage door serves a dual function. Located on the right side of the fuselage, over the leading edge of the wing (Figure 1-14), it can be used as an emergency exit and as an access to the aft cabin baggage

Figure 1-14. Emergency Exit/Baggage Door

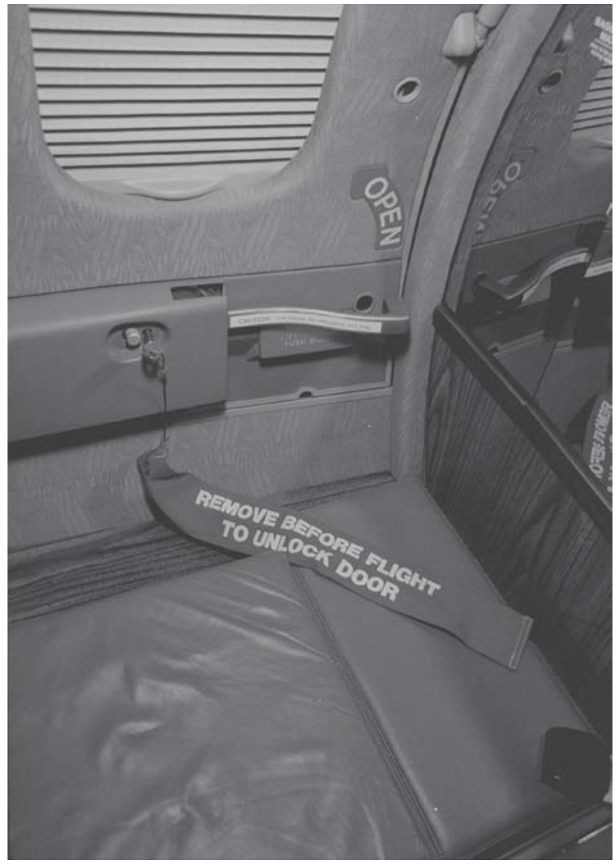


Figure 1-15. Emergency Exit/Baggage Door—Interior

area. There are latching handles inside and outside. The latching handle, when closed, drives six pins (three forward and three aft) into the sides of the doorframe and two pins into flanges on the lower doorframe. This door can be pinned from the inside for security purposes (Figure 1-15), but to facilitate emergency exit and ground emergency entrance, the pin must not be installed during flight.

The door can be opened from the inside or outside. The door is hinged at the top and has a torsion bar that assists opening and closing and holds the door in the open position. A damper is installed in the door to soften opening and closing door movements. When closed and the latching pins make contact, the AFT CAB DOOR light on the glareshield (Figure 1-12) goes out.

In an arrangement similar to the cabin entry door warning light arrangement, this is a steady light when the door handle is open or unlatched; it will flash if the handle is latched and the latching pins are not all properly engaged, or if the security pin is not removed.

The door contains a window similar to the other cabin windows, which will be discussed later.

Care should be taken that baggage is not dragged across the seal at the bottom of the frame. A protective fiberglass cap is provided with aircraft equipment. It should be set in place on the lower door frame to protect the door seal during baggage loading/unloading.

Windows

Windshield

The windshield consists of the pilot and copilot windshield halves (Figure 1-16). It is made of multiple layers of impact resistant acrylic plastic and is approximately one inch thick. The windshield has an integral electrical heating element for internal defogging.

Figure 1-16. Windshield and Window Locations

Cockpit Side Windows

Each side of the cockpit has a side window just aft of the windshield. The pilot side window (60-001 through 60-235) can be opened for direct communication with ground personnel or for ventilation (Figure 1-17). It hinges at the aft side and swings in. It is unlatched by two thumb latches at the forward side of the window. There is a sight port in the window frame panel, adjacent to the upper thumb latch, to allow a visual check of the locking operation. The crew side windows are laminated Plexiglas.

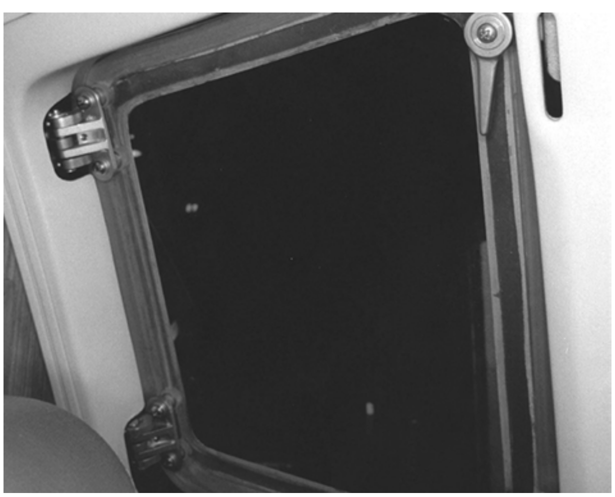


Figure 1-17. Pilot Side Window

Cabin Windows

There are eleven cabin windows, five on the left side and six on the right. They are made of two separate panes of acrylic plastic with dead air space in between. The aftmost window on the right-hand side is located in the emergency exit/baggage door. Each window has an accordion type shade installed.

Fuel Section

The fuel section is located aft of the rear pressure bulkhead. This section contains the fuselage fuel tank.

Tailcone Section

The tailcone section extends aft from the fuselage tank to the empennage. Two access doors (Figure 1-18) are on the left side of the tailcone. The forward door provides access to the tailcone baggage compartment, which can hold up to 300 pounds of baggage. The aft door provides access to the tailcone equipment area. Both doors are hinged at the bottom. The tailcone baggage door has a latching handle. The handle, when rotated closed, extends two latching pins into the door frame (one

Figure 1-18. Tailcone Baggage and Equipment Access Doors

on each side). Each pin contacts a microswitch that is connected to the EXT DOORS warning light. The tailcone access door handle extends two latching pins into the doorframe when in the closed position. These pins also contact microswitches. If any of the microswitches for the tailcone access door or the tailcone baggage door is not properly contacted, the red EXT DOORS light on the main annunciator panel illuminates (steady).

There is a light switch for the tailcone equipment compartment. This switch receives electrical power from the left battery bus. If inadvertently left on, it will be turned off by the door-closing action.

WING

The Learjet 60 has a sweptback, cantilevered, all-metal wing (Figures 1-19 and 1-20) that is mounted to the lower fuselage and joined together at the centerline of the fuselage.

Most of the wing is sealed to form an integral fuel tank.

Some of the significant items associated with the wing are:

- Winglets at the outboard end of each wing
- Two full-chord stall fences are on each wing, one on either side of the aileron. Between the fences there are two rows (10 each) of boundary layer energizers (BLEs).
- A triangle strip is installed on the inboard section of each wing leading edge.
- The winglets convert drag, induced by wingtip vortices, into a net forward thrust vector.

Figure 1-19. Learjet 60 Wing

Stall fences channel airflow across the portion of the wing forward of the ailerons. They delay disruption of the airflow at high angles of attack.

There are two rows of 10 boundary layer energizers (BLEs) that are small, delta-shaped bars. They delay airflow separation at higher Mach numbers, preventing unwanted aileron activity within the operating limits of the aircraft.

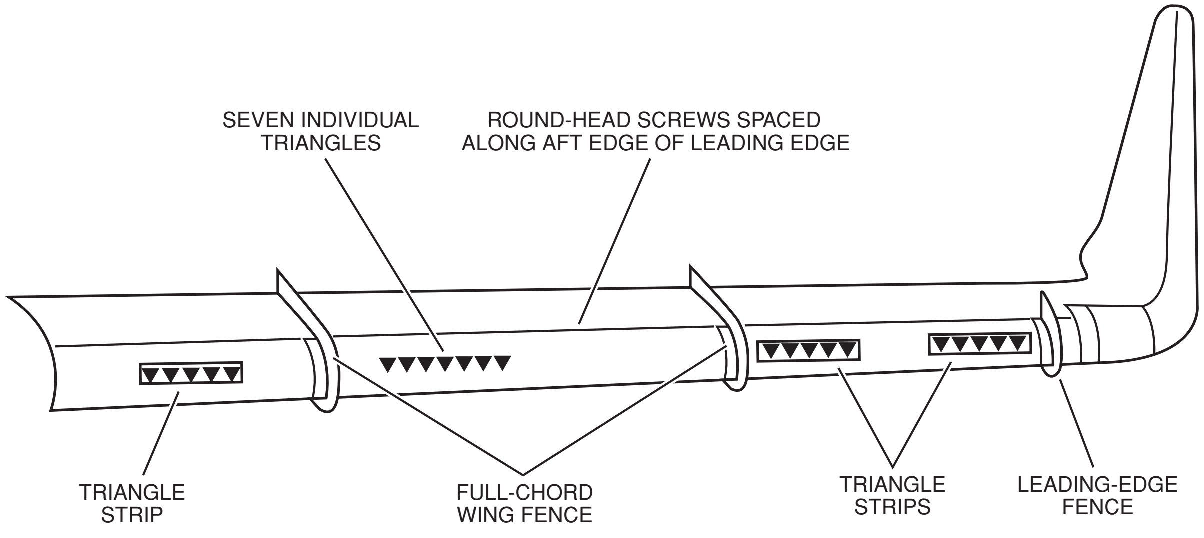


Figure 1-20. Wing Configuration

A triangle strip is affixed to the inboard section of each wing leading edge.

It generates a buffet at high angles of attack, which warns of an impending stall independent of the stall warning system. Its positioning is critical to the design flight characteristics and should be factory repaired if it becomes dislodged.

Seven triangles are applied to the leading edge of the wing between the fences (Figure 1-20). This modification improves the aircraft's stall characteristics.

Round-head screws are inserted at intervals along the aft side of the leading edge. These screws serve a dual purpose of attaching the leading-edge section to the wing and acting as BLEs at high angles of attack.

A leading-edge fence is located at the wingtip (Figure 1-20) just inboard of the navigation light. Two triangle strips are attached to the outboard section of the leading edge. The purpose of these triangle strips is to closely control stall onset on the various sections of the wing.

The wing contains conventional ailerons, single-slotted flaps, and spoilers, immediately forward of the flap, on the wing top surface. The main landing gear is attached to, and housed within, the wings.

EMPENNAGE

The T-tail empennage (Figure 1-21) includes a vertical stabilizer with an attached rudder and a horizontal stabilizer with connected elevators.

There is a dorsal fin that contains a ram air scoop at the front of the fin and an air scoop on each side of the fin. The side air scoops ventilate the tailcone. The vertical stabilizer has a 35.6° sweepback and is the mounting point for the rudder, and horizontal stabilizer, and the recognition light.

The horizontal stabilizer has a 25° sweep back and is attached to the vertical stabilizer at two points. The aft attached point serves as a pivot

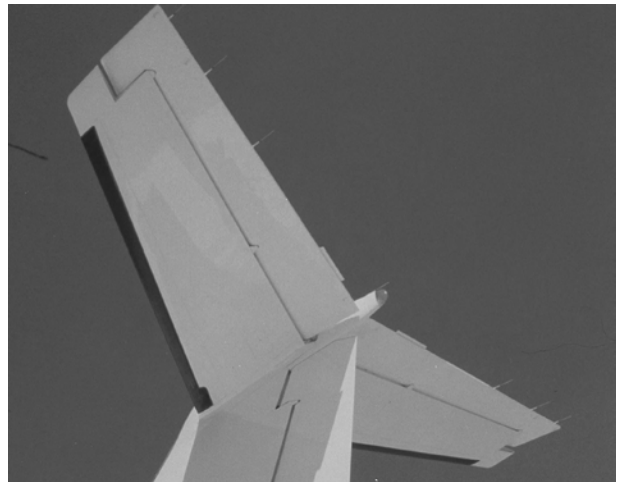


Figure 1-21. Empennage

for pitch trim adjustments. Elevators are mounted at the rear of the horizontal stabilizer.

The Learjet 60 model incorporates a pair of outward-canted ventral fins referred to as "delta fins." The delta fins are aluminum with honeycomb construction. The primary purpose of the delta fins is to prevent the deep stall typical of T-tail aircraft. Another benefit of the delta fins is that Dutch roll is damped out very quickly, improving directional stability. The aerodynamic effect of the delta fins has eliminated the need for a pusher system and allows the aircraft to be dispatched without an operating yaw damper.

STATIC DISCHARGE WICKS

There are four static discharge wicks on each winglet, two on each delta fin, three on each elevator and one on the tail navigation light housing for a total of 19 (Figure 1-22). See Appendix 1, Configuration Deviation List for Learjet Model 60 in the *LR60 Airplane Flight Manual* for limitations. The following five discharge wicks may be missing for flight:

- One on each winglet
- One on top of vertical tail navigation light
- One on each elevator (inboard most one)

The delta fins cannot have any discharge wicks missing.

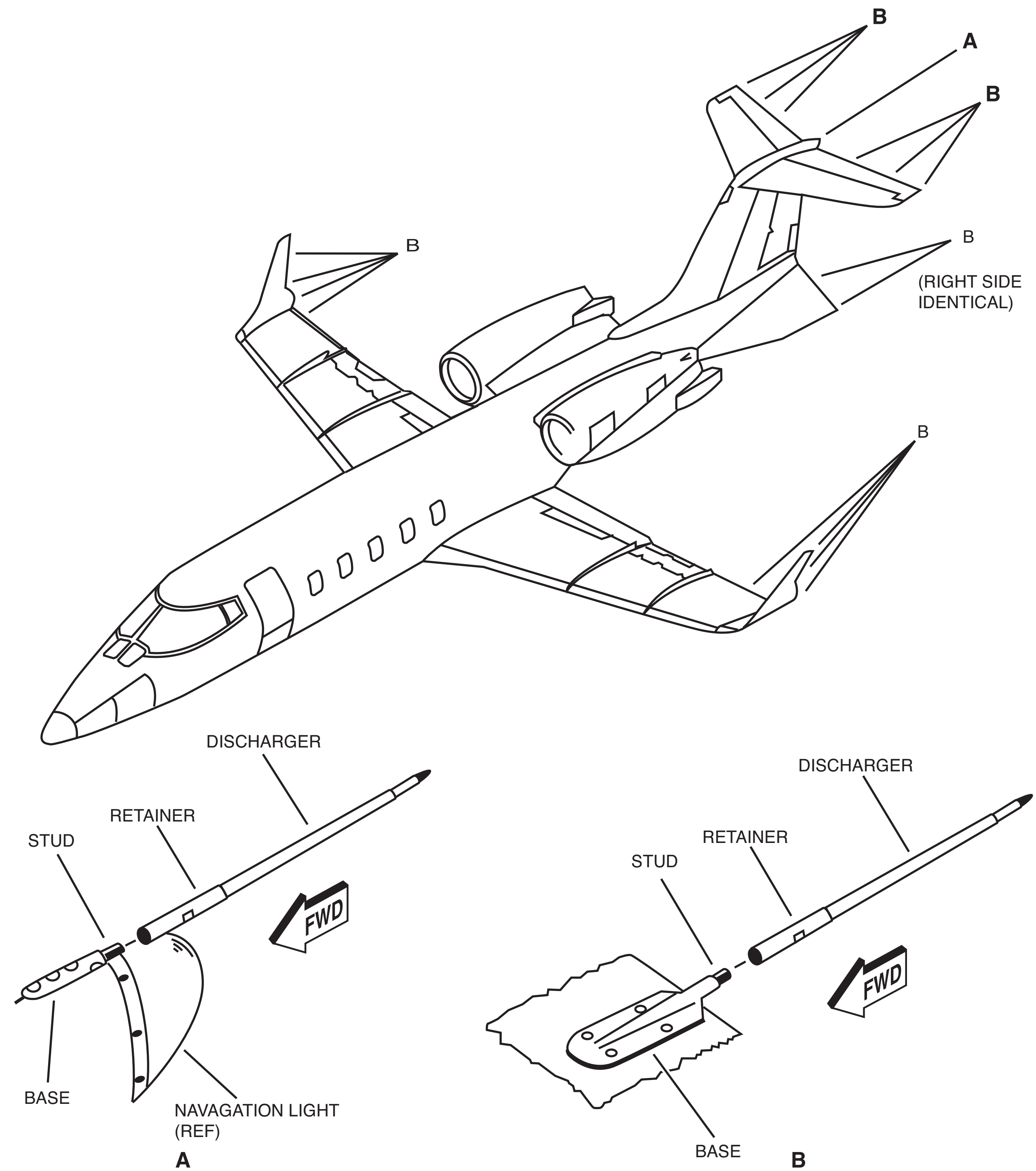


Figure 1-22. Static Discharge Wicks

AIRPLANE ANTENNAS

Figure 1-23 shows the location of the airplane antennas. Familiarity with antenna locations is an essential element of an external airplane inspection.

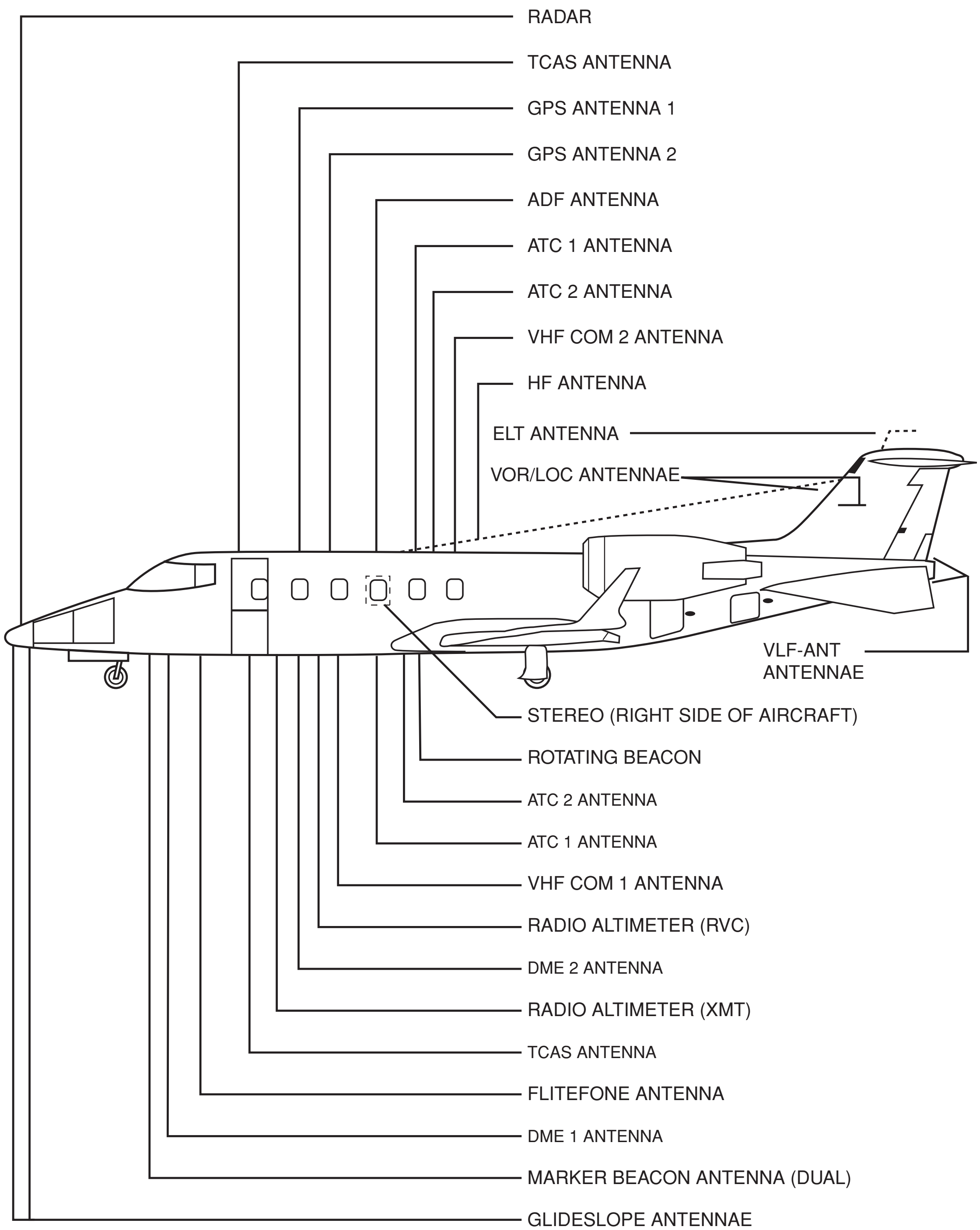


Figure 1-23. Aircraft Antenna Locator

CHAPTER 2
ELECTRICAL SYSTEMS

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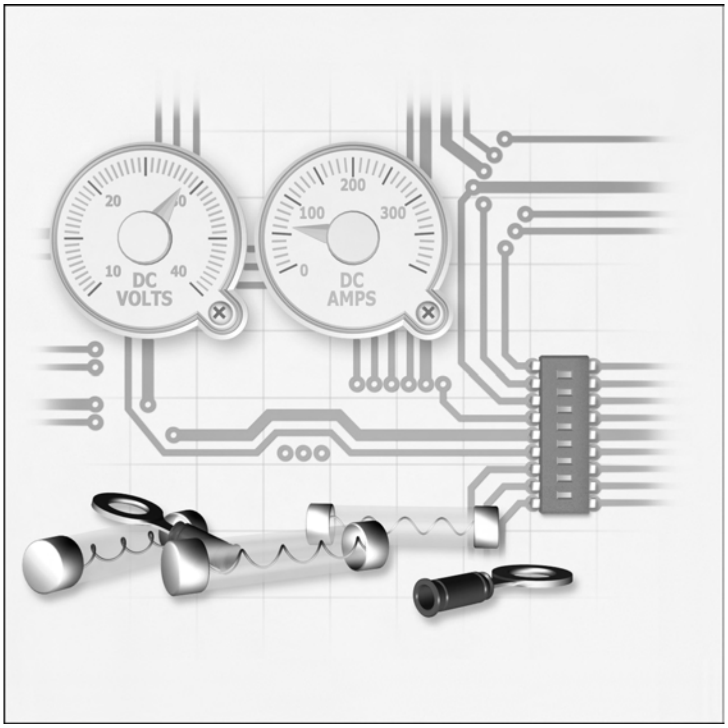
ILLUSTRATIONS

CHAPTER 2
ELECTRICAL SYSTEMS



INTRODUCTION

Primary DC electrical power is provided by two engine-driven starter/generators rated at 30 volts, 400 amperes each. A single starter/generator is capable of sustaining a normal DC load. Secondary DC electrical power is supplied by two main aircraft batteries. The batteries are capable of powering the entire electrical system and may be used to power emergency bus equipment, for a limited amount of time, if both generators become inoperative. A ground power unit can also provide electrical power for system operation or engine starting. Additionally, an optional APU may be installed to provide for systems operation on the ground and for engine start (see Chapter 6).

AC electrical power is provided by two solid-state inverters located in the tailcone. The inverters require DC power for operation.

Emergency batteries are provided in case of aircraft total electrical system failure.

GENERAL

The electrical system incorporates a multiple bus system for power distribution interconnected by relays, current limiters, overload sensors, and circuit breakers which react automatically to isolate a malfunctioning bus. Manual isolation is also possible by opening the appropriate circuit breakers.

In the event of a dual generator failure, the main aircraft batteries may be used to power the emergency bus system for approximately one hour if the Emergency Bus is selected. An emergency battery system is provided to operate selected equipment in the event of aircraft total electrical system failure.

It is possible to power the DC and AC electrical systems from the aircraft batteries, an engine-driven generator, ground power unit (GPU), or APU (if installed).

Figure 2-1 shows the major electrical power system component locations.

DC POWER

BATTERIES

Two batteries, one above the other, are in the tailcone (Figure 2-2). BAT 2 is above BAT 1. The standard batteries are 24 volt, 43 ampere-hour, lead-acid. Optional nickel-cadmium batteries may be installed. The batteries are vented overboard through a sump jar (lead-acid only) and tubes.

Each battery is connected to its battery bus, (hot-wired) through a current limiter for hot-wired circuits.

CONTROLS AND INDICATORS

The electrical switches and indicators on the Learjet 60 include an electrical switch panel containing the main battery switches. These are two-position switches, OFF and BATTERY 1 and OFF and BATTERY 2 (Figure 2-3). Generator reset buttons are just outboard of the battery switches. These buttons may be used to reset an engine-driven generator in the event of a malfunction. Three-position starter/generator switches are outboard of the generator reset buttons. These switches, START-OFF-L and R

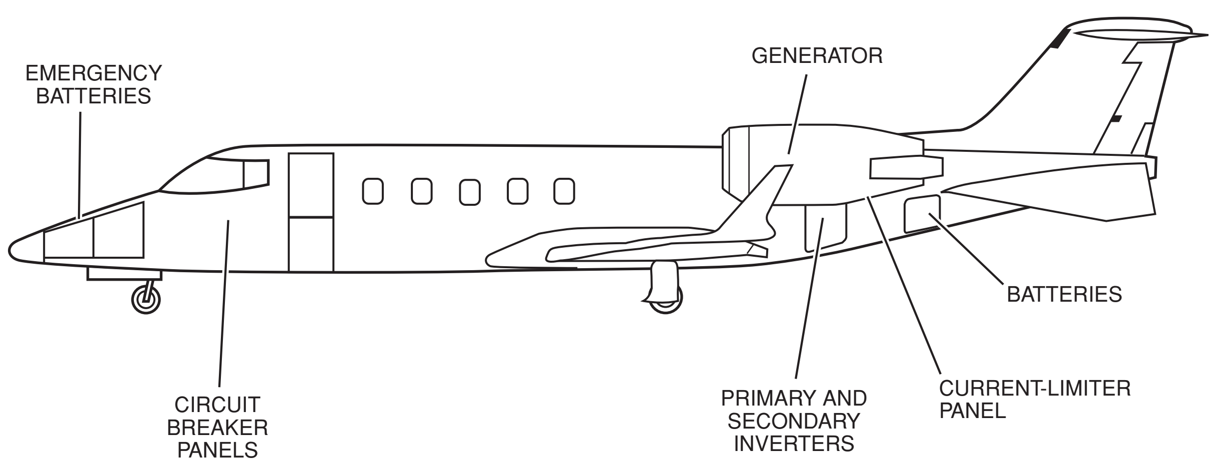


Figure 2-1. Component Locations

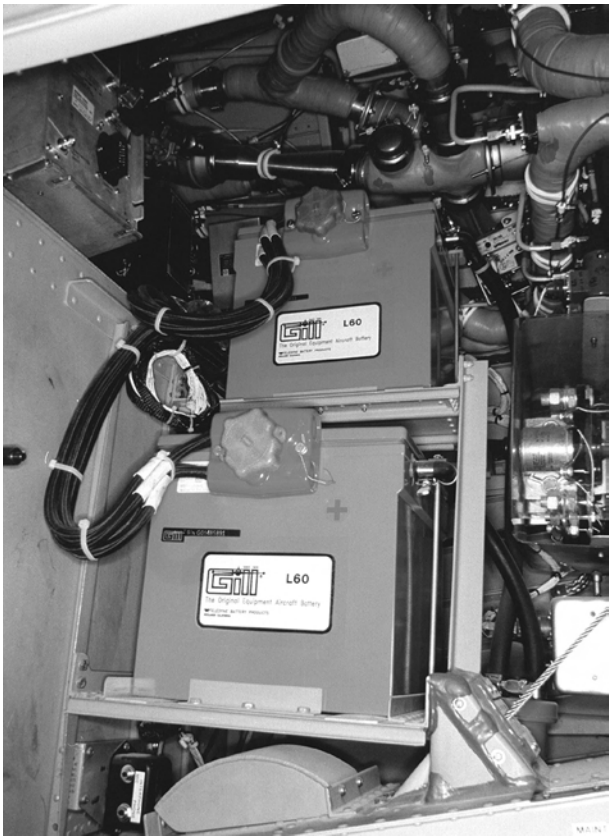


Figure 2-2. Battery Location

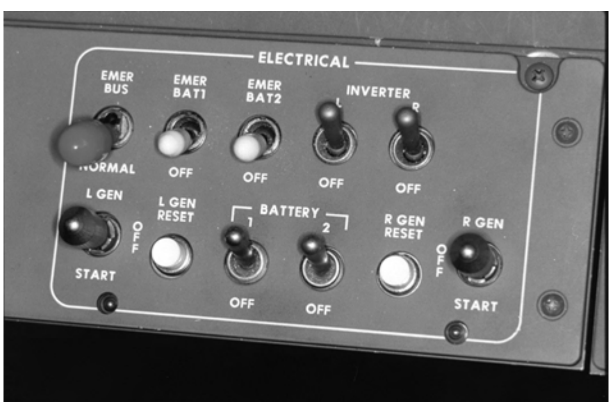


Figure 2-3. Battery Controls

GEN, are used to control the starter and generator functions of the engine-driven starter/generators. Amber lights that indicate the starter is engaged are just below the starter/generator switches.

One, two or three EMER BAT switches on the upper portion of the electrical switch panel are used to turn the emergency batteries ON and OFF. L and R INVERTER switches are to the right of the EMER BAT switches and are used to turn the inverters ON and OFF. A large, red, two-position switch in the upper left corner of the electrical switch panel is labeled NORMAL-EMER BUS. This switch is used to connect emergency bus equipment directly to the main aircraft batteries and shed nonessential loads in the event of dual generator failure.

An electrical power monitor (Figure 2-4) just above the electrical switch panel provides five digital displays. The upper two labeled VAC indicate the voltage on the left and right AC buses respectively. The display below them labeled VDC normally indicates the voltage on the battery charging bus. The VDC display indicates the highest voltage being applied to the bus from aircraft batteries, aircraft generators, a GPU, or APU (if installed). If, however, the EMER BUS-NORMAL switch is in the EMER BUS position, the VDC display indicates the highest battery voltage. The two bottom displays labeled AMP indicate the load in amperes being carried by the respective left and right generators. With both generators on, the meters should normally read within 40 amps of each other.

The electrical power monitor also has two annunciator lights, one amber and one red, which are used to alert the crew if any of the displays deviate from a preestablished range. If any display moves from the normal to a cautionary range (high or low), the affected display and the amber light on the electrical power monitor and the amber ELEC PWR glareshield annunciator light will all flash to attract the crew's attention. Both master caution lights will also flash. Depressing the amber light on the electrical power monitor or either master caution light will cancel the master caution and cause the display and other lights to stop flashing. The amber light on the electrical power monitor and the ELEC PWR annunciator light will remain illuminated steady as long as the malfunction remains.

If any display moves from the normal to an emergency range, the affected display, the red light on the electrical power monitor, and the amber ELEC PWR glareshield annunciator will all flash. Both master warning lights will

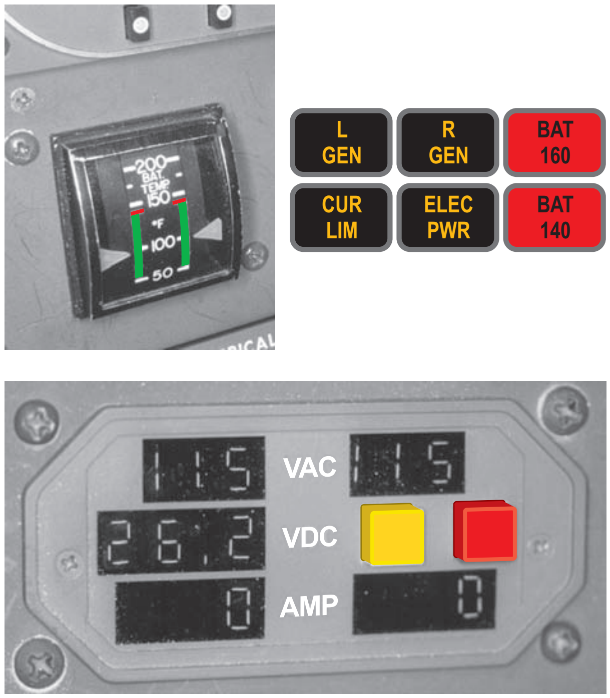


Figure 2-4. Electrical Power Monitor

also flash in this case. Depressing either master warning light will cancel the master warning lights and cause the display and other lights to stop flashing. The red light on the electrical power monitor and the ELEC PWR annunciator light will remain illuminated steady as long as the malfunction remains.

Except during system tests, the amber and red lights on the electrical power monitor will not normally be on simultaneously.

See Electrical Power Monitor in the Limitations section of the *Airplane Flight Manual* for the specific range for normal, cautionary, and emergency annunciation.

Amber glareshield annunciator lights are provided to monitor the generators. The lights labeled L and R GEN will be illuminated when the starter/generator switch is in START or OFF. The GEN lights will also be illuminated if a GPU is connected to the aircraft electrical system.

If, after engine start and GPU disconnect, a GEN light illuminates with the corresponding starter/generator switch in the GEN position, a malfunction is indicated.

A single amber glareshield annunciator light, labeled CUR LIM, monitors the continuity of two 275-amp current limiters which conduct current between the generator buses and the battery charging bus (Figure 2-5).

The light will illuminate if either, or both, 275-amp current limiters have failed.

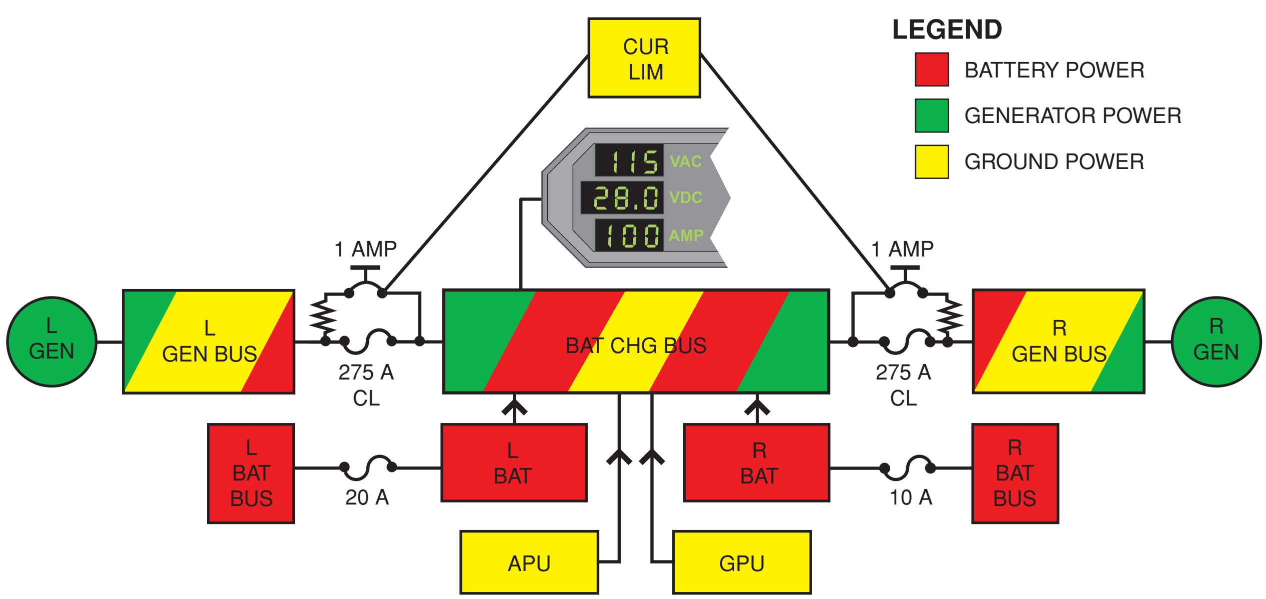


Figure 2-5. 275-Amp Current Limiters

Additionally, on aircraft with nickel-cadmium (nicad) batteries, two red annunciator lights (Figure 2-4), labeled BAT 140 and BAT 160, are provided to monitor the temperature of the main aircraft batteries. The lights will illuminate if either, or both, batteries overheat. To identify which battery has overheated, a dual-scale battery temperature indicator (Figure 2-4), above the electrical switch panel, indicates the temperature of the left and right main aircraft batteries individually.

GENERATORS

Two engine-driven starter/generators, one on each engine accessory section, provide the normal source of aircraft 28 volt DC power (Figure 2-6). Each generator is rated at 30 VDC, 400 amperes. After engine start, internal circuitry switches the starter to a generator at approximately 45% N_2. Cooling air is routed from a scoop on the associated engine nacelle to the associated starter/generator. During normal operation both generators operate in parallel through the generator control units. As long as the battery switches are on, either generator charges the batteries through the 275-amp current limiters. The generators supply DC power to all DC powered equipment on the aircraft.

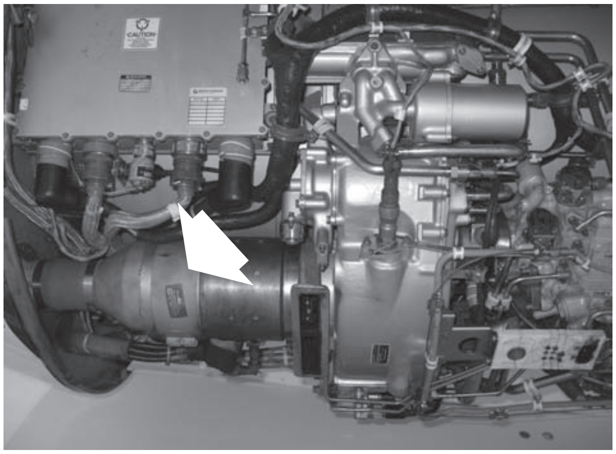


Figure 2-6. Generator Location

Automatic Load-Shedding System

An automatic electrical load-shedding system automatically reduces generator loading in the event of a single-generator failure. The system is only active during flight (weight not on wheels). Should either the L or R GEN light illuminate in flight, the following loads will automatically shut down to reduce the load on the operating generator:

- CABIN PWR BUS loads
- Freon cooling system
- Cockpit floorboard heater system (if installed)
- Baggage compartment heater system (if installed)

If the generator is brought back on-line, these loads will be regained.

Generator Control Units

Left and right generator control units (GCUs) are provided to monitor and control the engine-driven starter/generators. They regulate the voltage of the starter/generators to approximately 28 volts and limit the output of the generators to approximately 325 amps on the ground. Additionally, the GCUs will disconnect the generators if a malfunction occurs. The GCUs also provide several engine starting functions. See Chapter 7—"Powerplant," for additional information.

DISTRIBUTION SYSTEM COMPONENTS

Current Limiters

Various-sized current limiters are installed throughout the electrical system to provide circuit protection. A current limiter is similar to a slow-blow fuse in that it will carry more than its rated capacity for short periods of time. Extreme or prolonged overloading will cause a current limiter to fail, isolating a particular circuit, and precluding progressive failure of other electrical components. Current

limiters cannot be reset. When a current limiter has failed, it must be replaced. It should also be replaced if it shows discoloration or other signs of heating or overloading. Most of the current limiters are in a current- limiter panel above the tailcone baggage compartment (Figure 2-7).

Two types of current limiters are used. The lower amperage current limiters (50 amps or less) are red and have a pin that protrudes if the current limiter fails.

The higher amperage current limiters are made of a gray ceramic material with a small window that allows visual inspection of current limiter integrity.

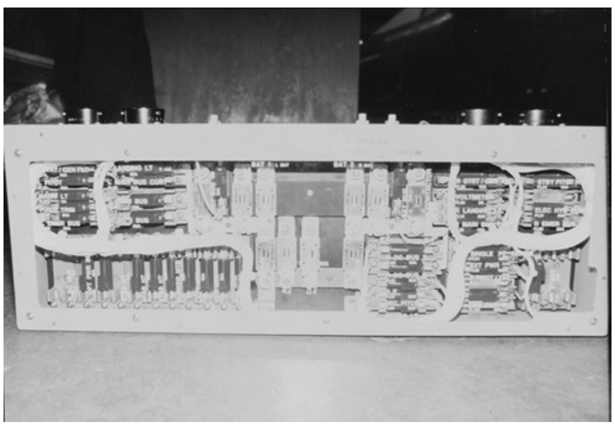


Figure 2-7. Current Limiter Panel

Relays

Relays are used throughout the electrical system, particularly in circuits with heavy electrical loads. Relays function as remote switches to make or break power circuits. Relays control the power circuits for the batteries, GPU, starter/generators, inverters, left and right DC 1 buses, and the cabin power bus.

Overload Sensors

Overload sensors and relays are used in the power circuits to the left and right DC 1 buses and in the power circuit to the cabin power bus. These sensors react thermally to electrical overloads. An overload sensor is a thermal-type mechanism that heats up and trips when a load in excess of its rated value passes through it. When the overload sensor trips, it provides a ground to trip the associated control circuit breaker. This causes the relay to open and break the power circuit. Once the overload condition has been removed, the overload sensor cools and resets automatically. However, the extended control circuit breaker in the cockpit must be manually reset to restore power to the system.

Circuit Breakers

A circuit breaker is designed to open and interrupt current flow in the event of a malfunction. Once opened, it may be reset by pushing it back in, but if it opens again, do not reset. Wait one minute before resetting a thermal circuit breaker to allow sufficient time for cooling. An open circuit breaker may be identified by the white base of the circuit breaker that can be seen only when the circuit breaker is open.

Most of the aircraft's circuit breakers are on two circuit-breaker panels in the cockpit, one left of the pilot seat and one right of the copilot seat (Figure 2-8). The copilot circuit-breaker panel also contains bus tie circuit breaker switches, explained later in this chapter.

The DC circuit breakers are thermal and the AC circuit breakers are magnetic. Amperage ratings are stamped on the top of each circuit breaker.

Figures 2-9 and 2-10 show typical circuit-breaker panels.

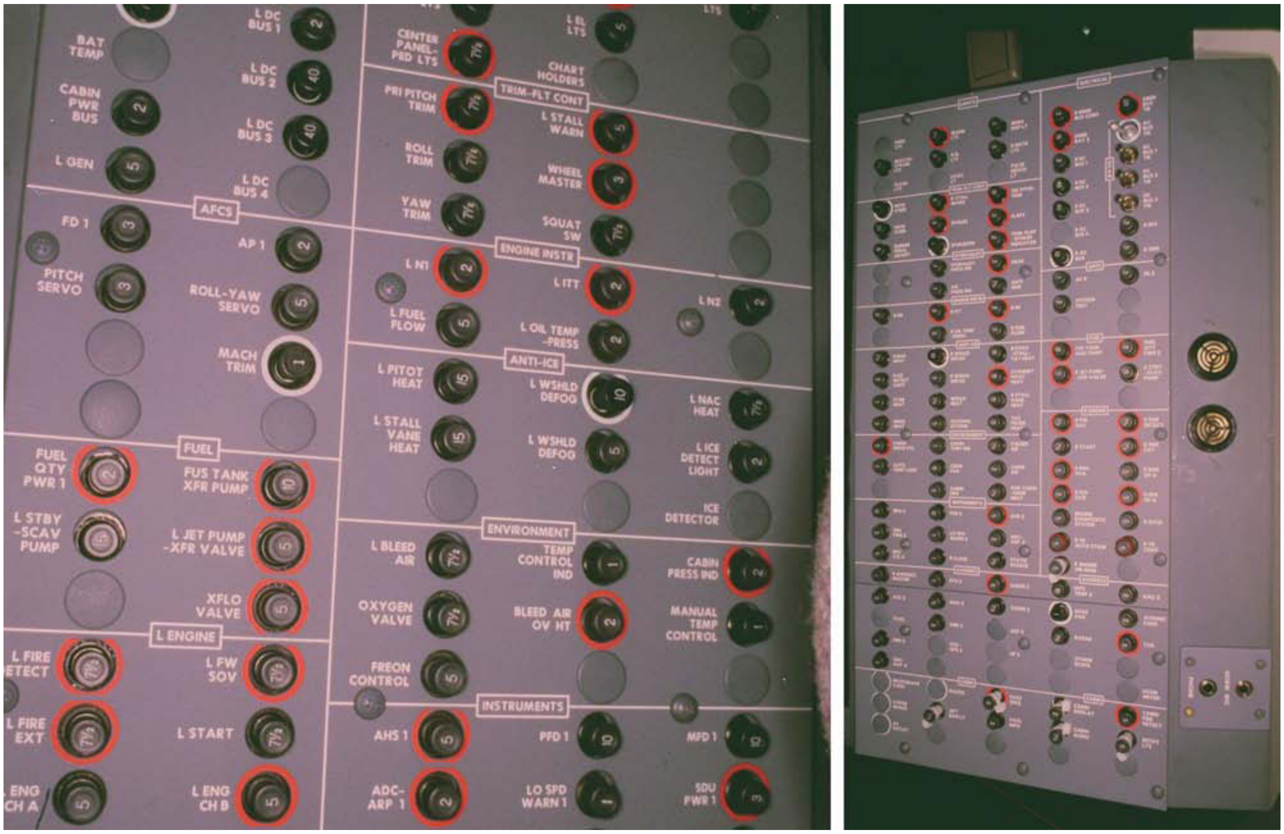


Figure 2-8. Pilot and Copilot Circuit-Breaker Panels

Circuit-Breaker Panels

On Learjet 60 aircraft, the circuit breakers are grouped by systems rather than buses (See Figures 2-9 and 2-10).

Emergency bus circuit breakers have red rings around them and AC circuit breakers have white rings around them, painted on the circuit-breaker panel.

All the bus tie circuit breaker switches are in the ELECTRICAL group on the copilot circuit-breaker panel. These switches are also circuit breakers and will physically move to the down (open) position if they have been raised (closed) and an overload occurs. These switches are normally left down in the open position. The bus tie circuit breaker between the left and right emergency buses is not a circuit-breaker switch.

The right bus circuit breakers, which connect the right buses to their power sources, are also in the ELECTRICAL group on the copilot circuit-breaker panel. The left bus circuit breakers, which connect the left buses to their power sources, are in the ELECTRICAL group on the pilot circuit-breaker panel.

DENOTES DC CIRCUIT BREAKERS

DENOTES AC CIRCUIT BREAKERS

DENOTES CIRCUIT BREAKERS ON THE EMERGENCY BUS

DENOTES UNUSED CIRCUIT BREAKER POSITIONS

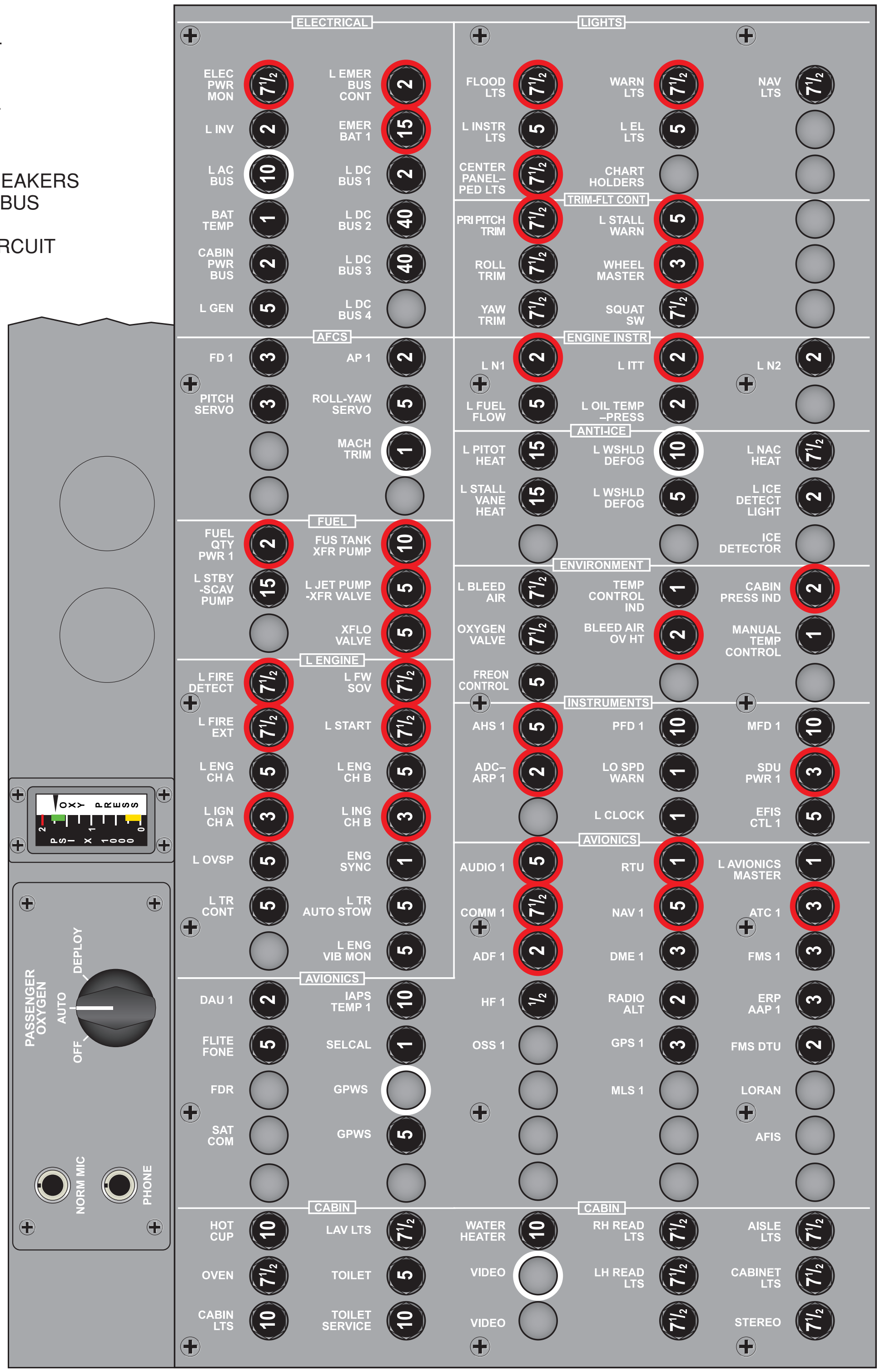


Figure 2-9. Pilot Circuit-Breaker Panel (Typical)

- DENOTES DC CIRCUIT BREAKERS
- DENOTES AC CIRCUIT BREAKERS
- DENOTES CIRCUIT BREAKERS ON THE EMERGENCY BUS
- DENOTES UNUSED CIRCUIT BREAKER POSITIONS

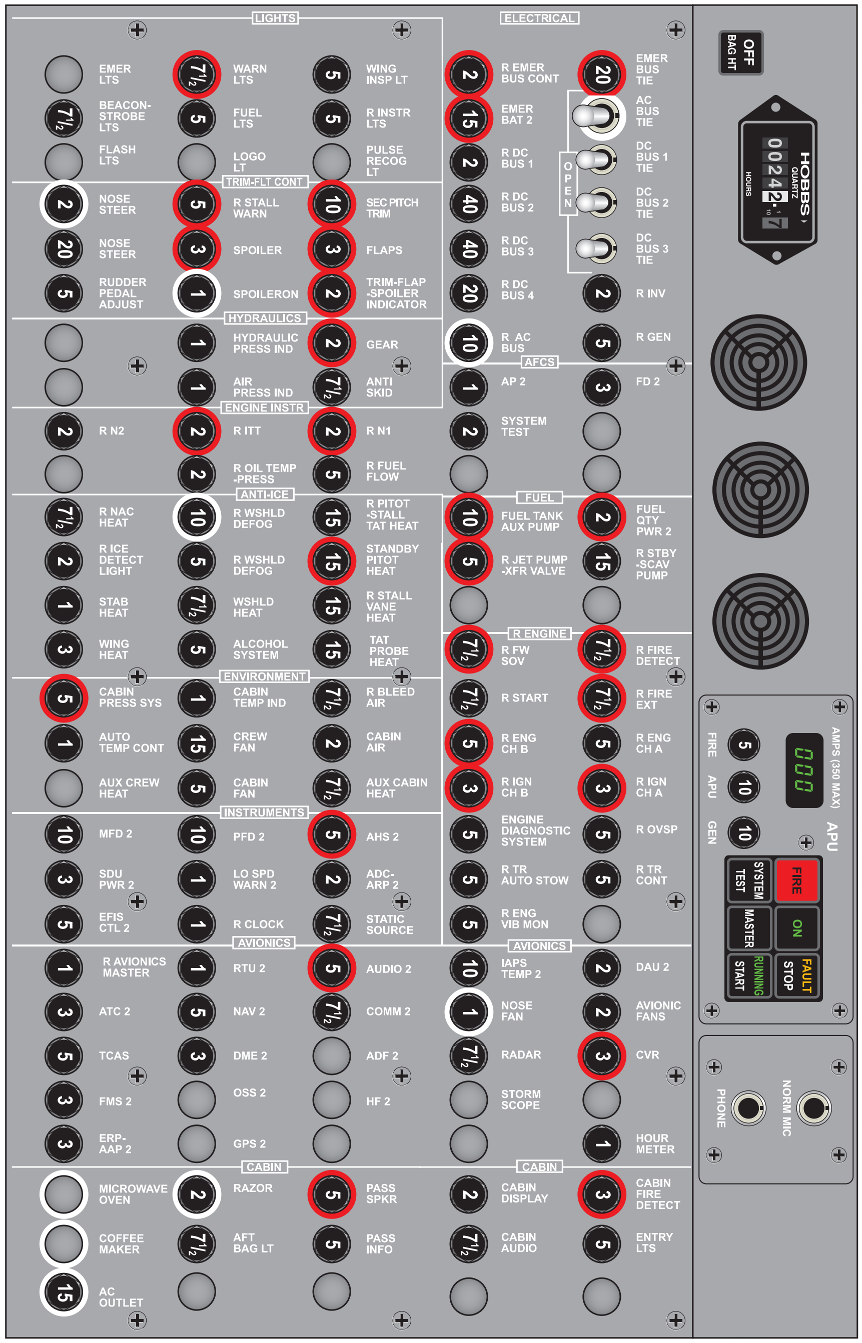


Figure 2-10. Copilot Circuit-Breaker Panel (Typical)

DISTRIBUTION

The aircraft basic DC power sources and distribution are shown in Figure 2-11. With the main aircraft batteries installed, power from the No. 1 battery, through a 20 amp current limiter, is immediately available to the hot wired items through the BAT 1 bus. These include the tailcone inspection light and toilet service receptacle. From the No. 2 main battery, through a 10 amp current limiter, power is immediately available from the BAT 2 bus to the cabin entry lights, cockpit dome lights (if remote selected and entry lights ON), the single point pressure refueling (SPPR) system, and to the aft cabin and tailcone baggage compartment lights.

All these items will operate, as long as the batteries are installed, even with the battery switches turned OFF.

Power from both batteries is also available through current limiters to the emergency bus system. However, as long as the red EMER BUS switch on the electrical switch panel is in NORMAL, these circuits are not powered directly from the batteries.

Each battery switch contains two contacts that close when the switch is turned on. One contact provides battery power to emergency bus relays; however,with the red EMER BUS switch in NORMAL, this circuit is open and the emergency bus relays are not powered. The other contact in the battery switch supplies battery power through a battery relay to another contact in the EMER BUS switch. With the EMER BUS switch in NORMAL, this contact provides a ground and the battery relay closes. When the battery relay closes, the battery is connected directly to the battery charging bus. The DC voltmeter is also connected to the battery charging bus. With one battery switch on, and the battery relay closed, that battery's voltage may be checked on the DC voltmeter (Figure 2-11).

The battery charging bus can be powered from the aircraft batteries, either or both generators, a ground power unit, or an auxiliary power unit (if installed). The battery charging bus acts as a DC distribution point for many items (Figure 2-12).

From the battery charging bus, through two, 275-amp current limiters, battery power is also applied to the left and right generator buses. These two current limiters are monitored by the single amber CUR LIM annunciator light mentioned previously.

It takes approximately 16 volts to close the battery relay. If a battery is discharged to the point where its voltage is too low to close the relay, the battery cannot be connected to the battery charging bus.

Once either battery switch is on and the battery relay is closed, power from either battery will close an external power control relay, which will allow a ground power unit or APU to be connected to the aircraft.

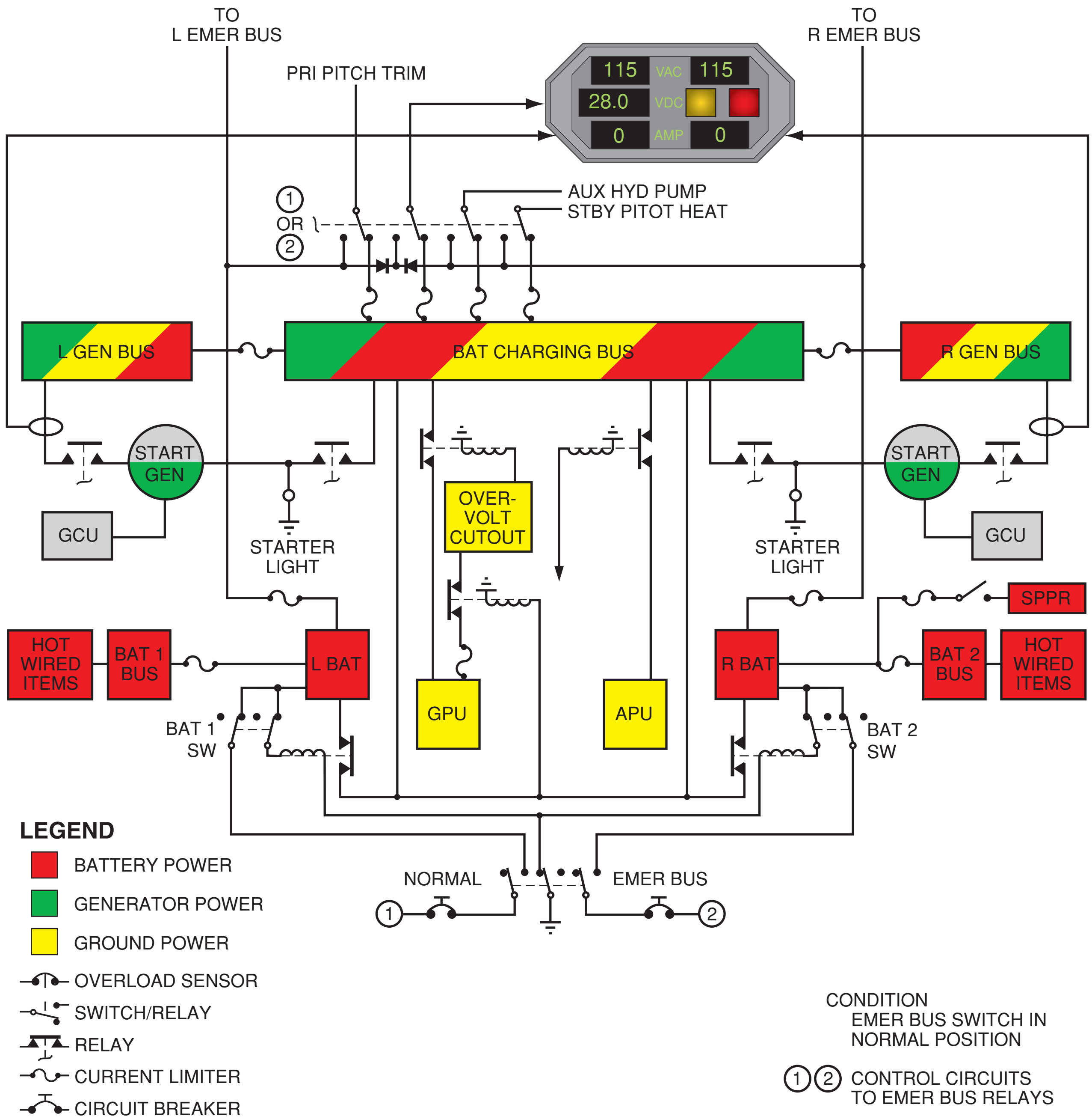


Figure 2-11. Basic DC Power Distribution

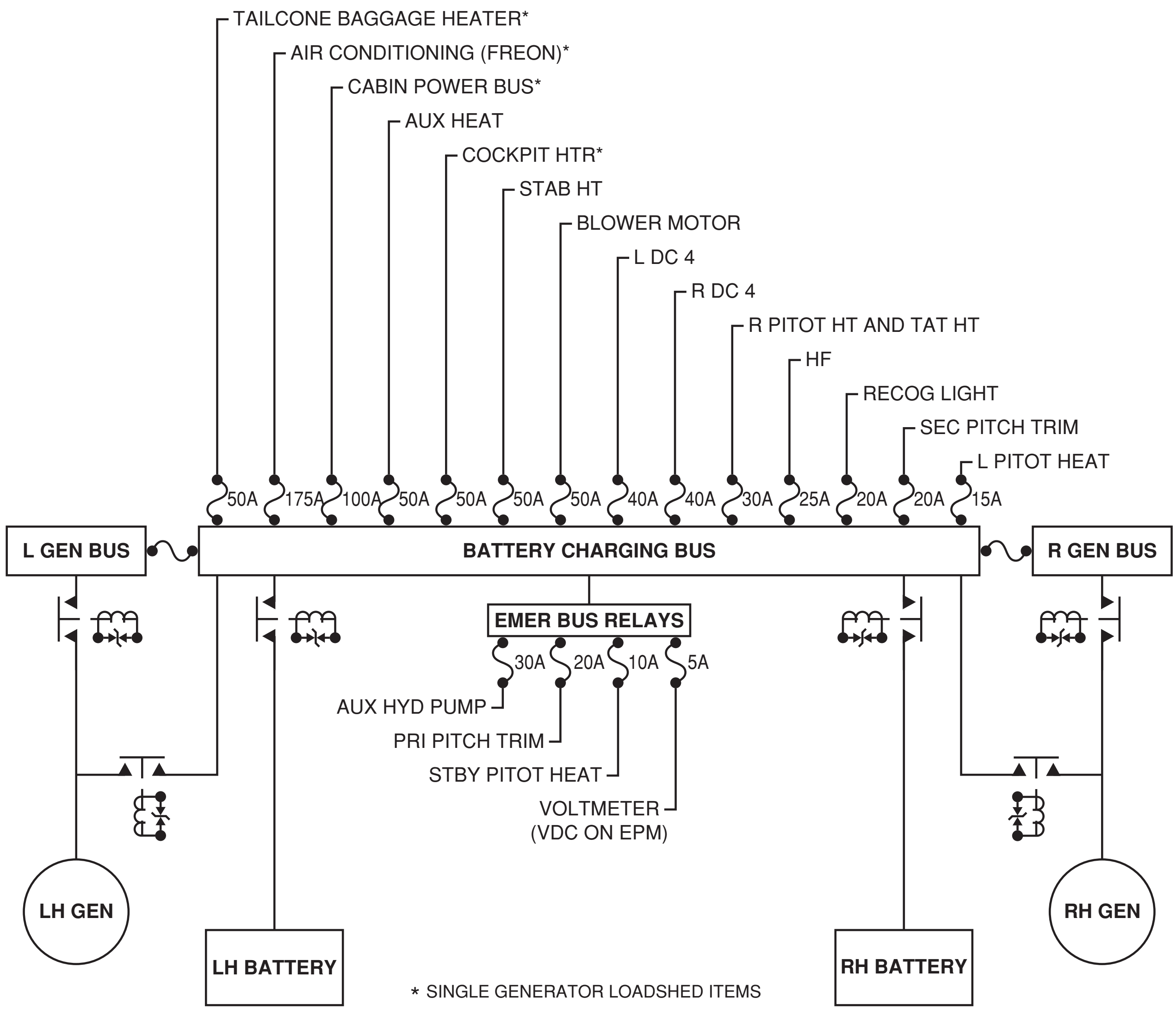


Figure 2-12. Battery Charging Bus Distribution

Ground power can be connected to the aircraft through a receptacle on the aft right side of the fuselage (Figure 2-13). With ground power connected, the output of the GPU is applied to the battery charging bus and through the two 275-amp current limiters to the left and right generator buses. GPU voltage will be indicated on the DC voltmeter.

The GPU should be regulated to 28 volts and limited to 1,500 amperes for engine starting. The GPU should be capable of producing at least 500 amps or the main aircraft batteries must pick up the load on start. An overvoltage cutout circuit in the ground power monitor box will disconnect the GPU from the aircraft if GPU voltage exceeds approximately 33 volts. The ground power monitor box is on the aft side of the generator control panel.

See Chapter 6 for information on the optional APU electrical distribution.

The *AFM* recommends that a GPU or APU be used for engine start when the ambient temperature is 32˚F (0˚C) or below.

With an engine running, placing the corresponding starter/generator switch to GEN will signal the generator control unit (GCU) to connect the generator to the aircraft electrical system. However, if ground power (GPU) is connected to the aircraft, left and right generator lock-out relays will prevent the generators from coming on-line. If ground power has been disconnected and the starter/generator switch is in the GEN position, the GCU will close a generator relay and connect the output power from the generator to the generator bus. The amber L or R GEN annunciator light will extinguish and generator output will be applied to the corresponding generator bus. The generator bus will then distribute the DC current to the corresponding inverter, to other DC buses, and to the battery charging bus through the 275-amp current limiter. From the battery charging bus, generator output can be used to recharge the batteries and to power the opposite generator bus through the other 275-amp current limiter.

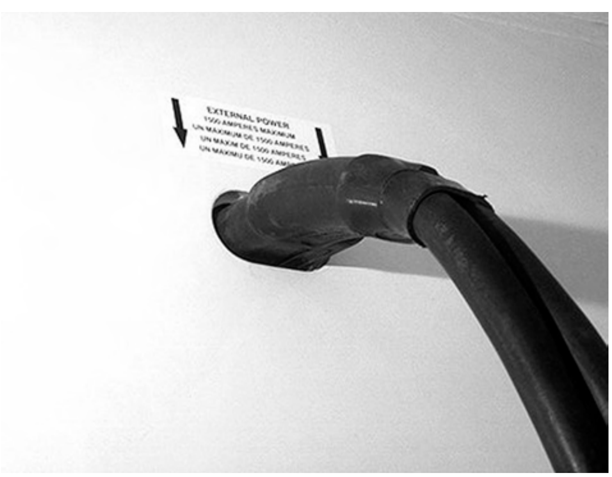


Figure 2-13. Ground Power Connector

With both 275-amp current limiters failed, both generator buses are disconnected from the battery charging bus. In this case, only battery voltage will be indicated on the DC voltmeter, even with both generators on and operating.

Cabin Power Bus

The CABIN PWR BUS is in the left circuit breaker panel and is shown in Figure 2-14. It is powered from the BATTERY CHARGING BUS through a 100-amp current limiter, a 70-amp overload sensor and a power relay. There is a CABIN PWR BUS control circuit breaker in the ELECTRICAL group of circuit breakers on the left circuit breaker panel that provides current to close the power relay.

DC 4 Buses

The DC 4 buses are powered from the BATTERY CHARGING BUS through 40-amp current limiters and 20-amp circuit breakers. These buses contain circuit breakers for optional avionics equipment (Figure 2-14). Not all aircraft have the DC 4 buses installed.

DC 2 and 3 Buses

The DC 2 and 3 buses are shown in Figure 2-14. They are powered from their respective generator buses through 50-amp current limiters and 40 amp circuit breakers. The left and right buses are normally powered separately. However, in the event of a malfunction, the

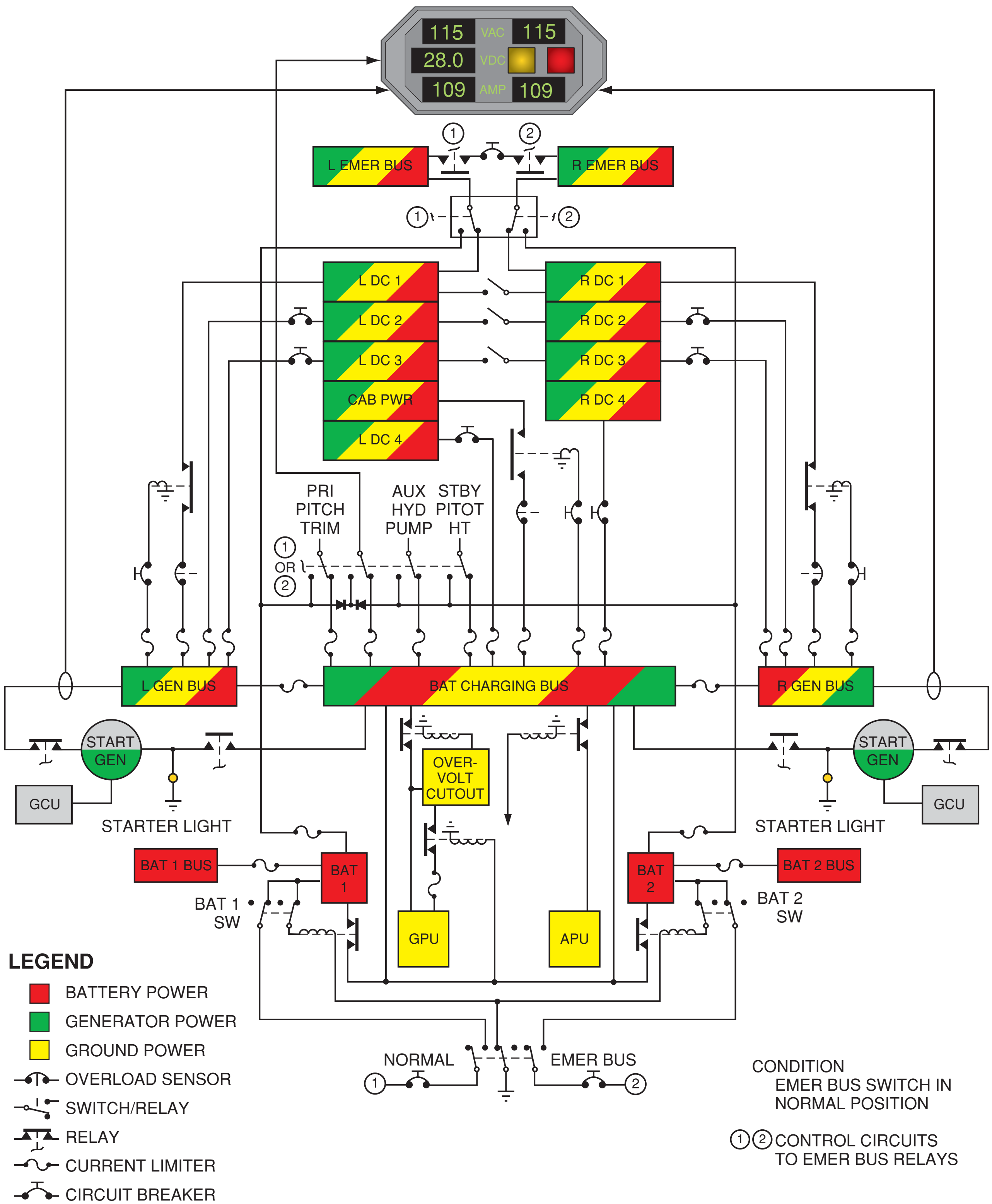


Figure 2-14. Normal DC Power Distribution

buses may be connected to each other through bus tie switches on the copilot circuit-breaker panel. These switches, which are also 20-amp circuit breakers, are normally in the down or open position. When the switch is raised, it closes and connects the buses, which allows one bus to power the bus on the opposite side. If there is excess current flow between the buses, the bus tie switch/circuit breaker will physically move to the down position to separate the buses.

DC 1 and Emergency Buses

The DC 1 buses and emergency buses are shown in Figure 2-14. The left and right DC 1 buses are powered from their respective generator buses through 70-amp overload sensors and power relays. Two-amp control circuit breakers, for the relays, are also powered from the generator buses, through 10-amp current limiters. The left and right DC 1 buses are normally powered separately. However, in the event of a malfunction, the buses may be connected to each other through a bus tie circuit-breaker switch on the copilot's circuit-breaker panel.

This bus tie switch, which is also a 50-amp circuit breaker, is normally in the down or open position. When the switch is raised, it closes, connecting the buses, allowing one bus to power the bus on the opposite side. If there is excess current flow between the buses, the bus tie switch/circuit breaker will physically move to the down position to separate the buses.

Power from each DC 1 bus is also applied to an emergency bus relay (EBR). With the red EMER BUS switch in NORMAL, power through these relays is then applied to the left and right emergency buses. In this configuration, the emergency buses are essentially part of the DC 1 buses on each side.

When the red EMER BUS switch is positioned to EMER BUS, as will be explained later, the emergency bus relays reposition. In this configuration, the emergency buses are disconnected from the DC 1 buses, and are powered directly from the respective batteries. They are also tied together through a 20-amp bus tie circuit breaker and two relays. Due to the two relays between the emergency buses, which are open with the EMER BUS switch in NORMAL, the emergency buses cannot be tied together unless the EMER BUS switch is in the EMER BUS position. See section later in this chapter, for additional information on the emergency buses.

AC POWER

Two 1500 VA inverters are installed in the tailcone equipment section above the baggage compartment to supply power to the AC distribution system. The left inverter supplies 115-VAC electrical power to the left AC distribution bus in the pilot circuit-breaker panel. The right inverter supplies 115-VAC electrical power to the right AC distribution bus in the copilot circuit-breaker panel (Figure 2-15). The left and right inverters also power the windshield defog system with 115-VAC (see Chapter 10, Ice and Rain Protection).

The L and R inverters are each supplied 28 VDC from the respective GEN BUS and are capable of supplying full output power with input voltages varying between 24 and 32 VDC. Each inverter provides an output of 115-VAC, 400 Hz.

Inverter relays complete the circuit between the inverters and the AC power system and are controlled by the left and right inverter switches on the ELECTRICAL switch panel.

A 100-amp current limiter and an inverter power relay is between the GEN BUS and the inverter, on each side, to control DC power to the inverters. The 115-VAC output from the left and right inverters is fed to the corresponding L and R AC BUS through a 10-amp current limiter, contact in the isolation relay and a 10-amp L or R AC BUS circuit breaker. A parallel output from each inverter goes through a 10-amp current limiter, a second contact in the isolation relay, and a 10-amp L or R WSHLD DEFOG circuit breaker for the windshield defog system.

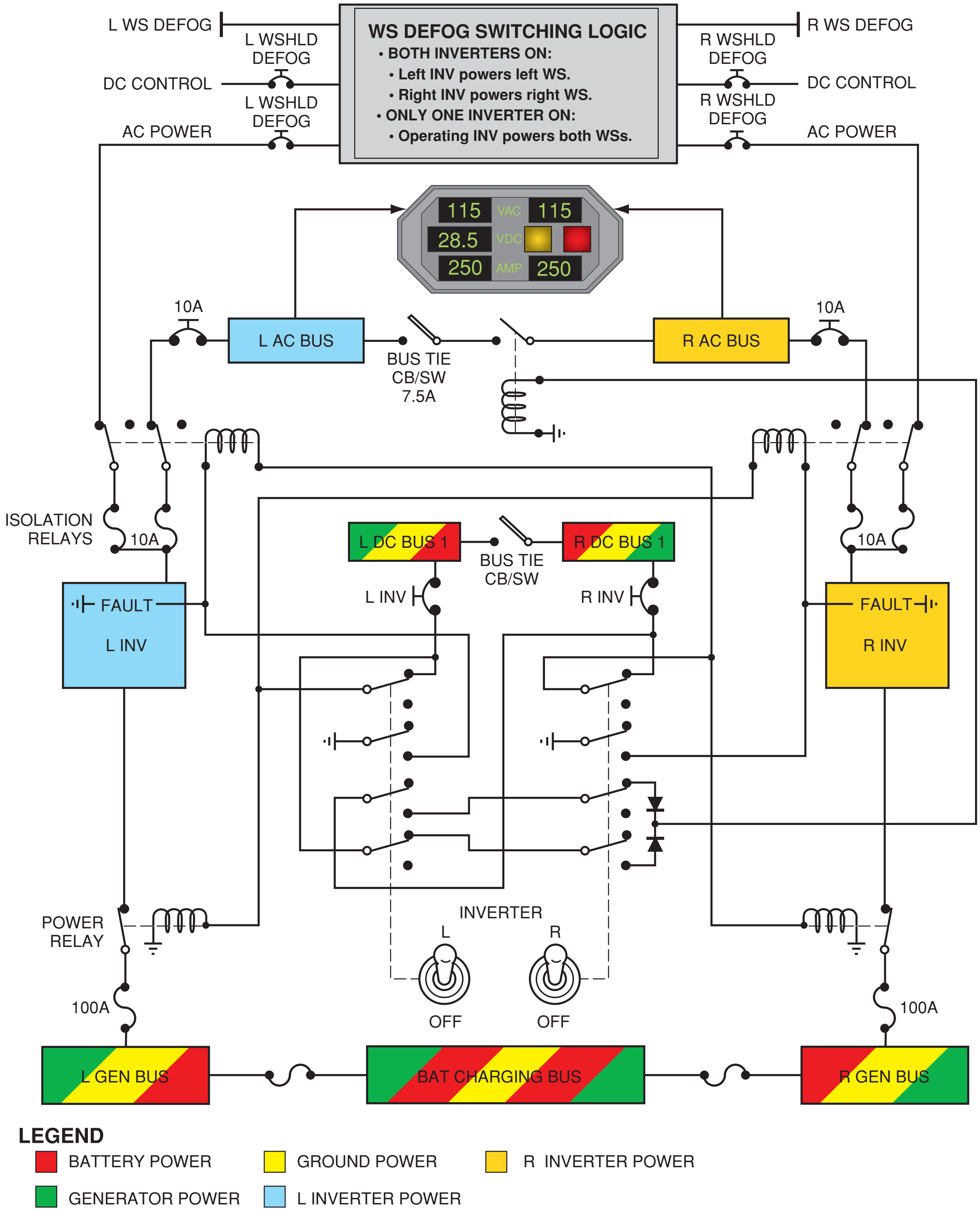


Figure 2-15. AC Distribution

CONTROLS AND INDICATORS

Two inverter switches, labeled L and R, on the electrical switch panel, control the left and right inverters (Figure 2-16).

Figure 2-16. Inverter Controls

When the L and R INVERTER switches are turned ON, the inverter power relays are energized closed, providing 28 VDC to the inverters. The current that energizes these relays closed comes from the L and R DC BUS 1 respectively, through the L and R INV circuit breakers, and then through contacts in the L and R INVERTER switches. With both inverters turned ON, the 115-VAC output of the inverters is fed through the corresponding isolation relays, which are deenergized closed, to each AC bus. The two AC buses are not normally connected to each other.

An AC bus tie circuit breaker switch is on the copilot circuit breaker panel. It may be used to tie the L and R AC distribution buses together when only one inverter switch is ON. There is also a bus tie relay in the circuit between the two AC buses that prevents the buses from being tied together when both inverters are ON. In order to tie the two AC buses together, one inverter switch must be OFF and the AC BUS TIE circuit breaker switch must be closed (up).

The electrical power monitor, as described earlier in this chapter under DC Power Control and Indicators, continuously monitors and displays the AC voltage on the L and R AC buses simultaneously. If voltage drops or exceeds the caution parameters, the digital readout and amber light on the electrical power monitor (see Figure 2-4) and the amber ELEC PWR annunciator on the glareshield will flash simultaneously. Both master caution lights will also flash.

If the voltage continues to increase or decrease and reaches the emergency parameters, the red light on the electrical power monitor will flash simultaneously with the digital VAC indicator and the amber ELEC PWR annunciator. The master warning lights will also flash.

SINGLE-INVERTER OPERATION

Setting the left inverter switch to ON with the right inverter switch OFF energizes the left inverter power relay closed and energizes the AC bus tie relay closed. With the right inverter switch off, the right inverter isolation relay is energized open. DC power is supplied to the left inverter, and the AC output of the inverter is supplied to the L AC BUS through the deenergized (closed) left isolation relay and the L AC BUS circuit breaker. With the right inverter OFF and the AC bus tie circuit breaker switch OPEN, power will be available to the L AC BUS only. Moving the bus tie circuit breaker switch to the closed (up) position will allow the L inverter to also power the R AC BUS through the bus tie circuit breaker switch and the bus tie relay.

The right inverter can supply AC power to both AC buses in the same manner by placing the right inverter switch ON and the left inverter switch OFF.

DUAL-INVERTER OPERATION

When both L and R inverter switches are ON, the inverters supply AC power to their respective buses and windshield defog systems.

The bus tie relay is deenergized open with both inverter switches ON and the buses cannot be tied together even if the AC bus tie circuit breaker switch is closed.

When the inverters are operating simultaneously, the left inverter supplies a signal to the right inverter which contains a phase-lock circuit to insure the right inverter remains in-phase with the left inverter. Should the inverters get out of phase, the right inverter sends a signal to the VAC display on the electrical power monitor and causes the last digit of both VAC displays to flash the letter C. The remaining digits will operate normally.

If either inverter should fault when both inverters are ON, the inverter with the internal fault provides a ground for the corresponding AC isolation relay circuit. This would energize the isolation relay open and disconnect the inverter from its respective AC bus (see Figure 2-15).

EMERGENCY BUS SYSTEM

The emergency bus system is shown in Figure 2-17. If after dual generator failure, the red EMER BUS switch is positioned to EMER BUS, the ground is removed from the main aircraft battery relays. The relays open, disconnecting the batteries from the battery charging bus. At the same time, through two other contacts in the EMER BUS switch, the emergency bus relays are powered. When they are, the left and right emergency buses are connected directly to the left and right main aircraft batteries and the buses are tied together through a 20 amp bus tie circuit breaker, and two relays.

The primary pitch trim will be powered from the left main aircraft battery and the auxiliary hydraulic pump and standby pitot-static heat will be powered from the right main aircraft battery. The DC voltmeter will be connected to both main aircraft batteries and will read the highest voltage of the two (Figure 2-17).

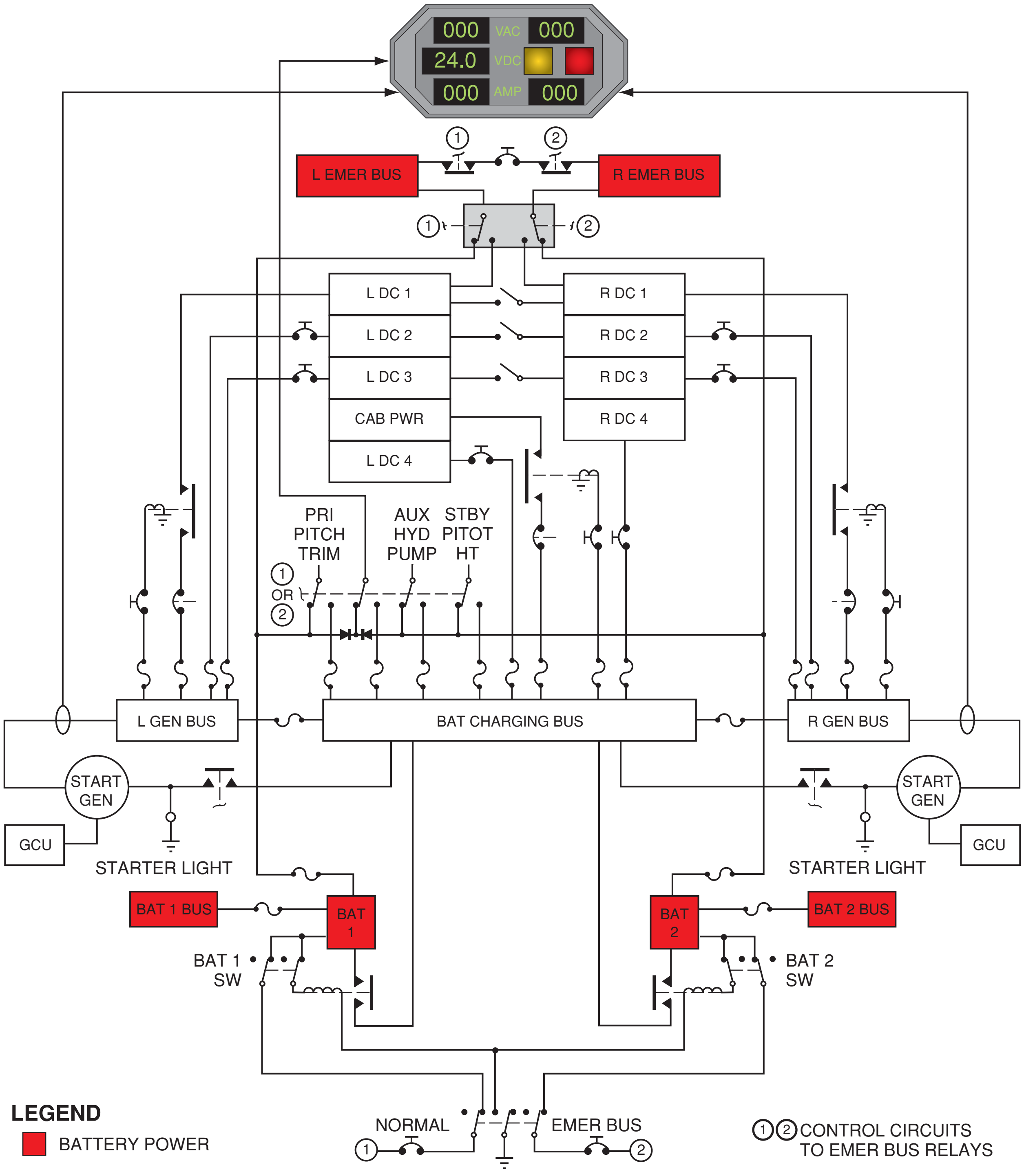


Figure 2-17. Emergency Bus Power Distribution

EMERGENCY BATTERIES

Learjet 60 aircraft are equipped with either one, two or three 24-volt, 5-ampere hour, lead-acid emergency batteries in the nose compartment (Figure 2-18). They provide an emergency electrical power source for selected equipment in the event of total aircraft electrical system failure (both generators and both main batteries). Each emergency power supply unit contains a 12-cell lead-acid battery to provide electrical power.

Figure 2-18. Emergency Battery Location

With normal electrical power on the aircraft, the emergency batteries receive a trickle-charge through 15 amp EMER BAT 1 and 2 circuit breakers on the left and right emergency DC buses (Figure 2-19). There are one, two, or three amber EMER PWR annunciator lights on the instrument panel above the standby attitude indicator. The lights will illuminate when power from the emergency batteries is being used and the emergency batteries are not receiving a trickle-charge from the aircraft electrical system.

On standard Learjet 60 aircraft with two emergency batteries and with the EMER BAT 1 switch ON, EMER BAT 1 will provide electrical power to the following equipment if the normal electrical power source for the equipment is lost:

- Fan speed (N_1) indicators
- Standby attitude indicator
- Instrument lighting for:
  - —fan speed (N_1) indicators
  - —magnetic compass
  - —standby attitude indicator
  - —standby airspeed indicator
  - —standby altimeter

With the EMER BAT 2 switch ON, EMER BAT 2 will provide electrical power to the following equipment if the normal electrical power source for this equipment is lost:

- Landing gear SAFE and UNSAFE lights
- AHS Computer No. 1 (for 11 minutes)
- AHS Computer No. 2 (for 11 minutes)
- ADC 1
- ADC 2

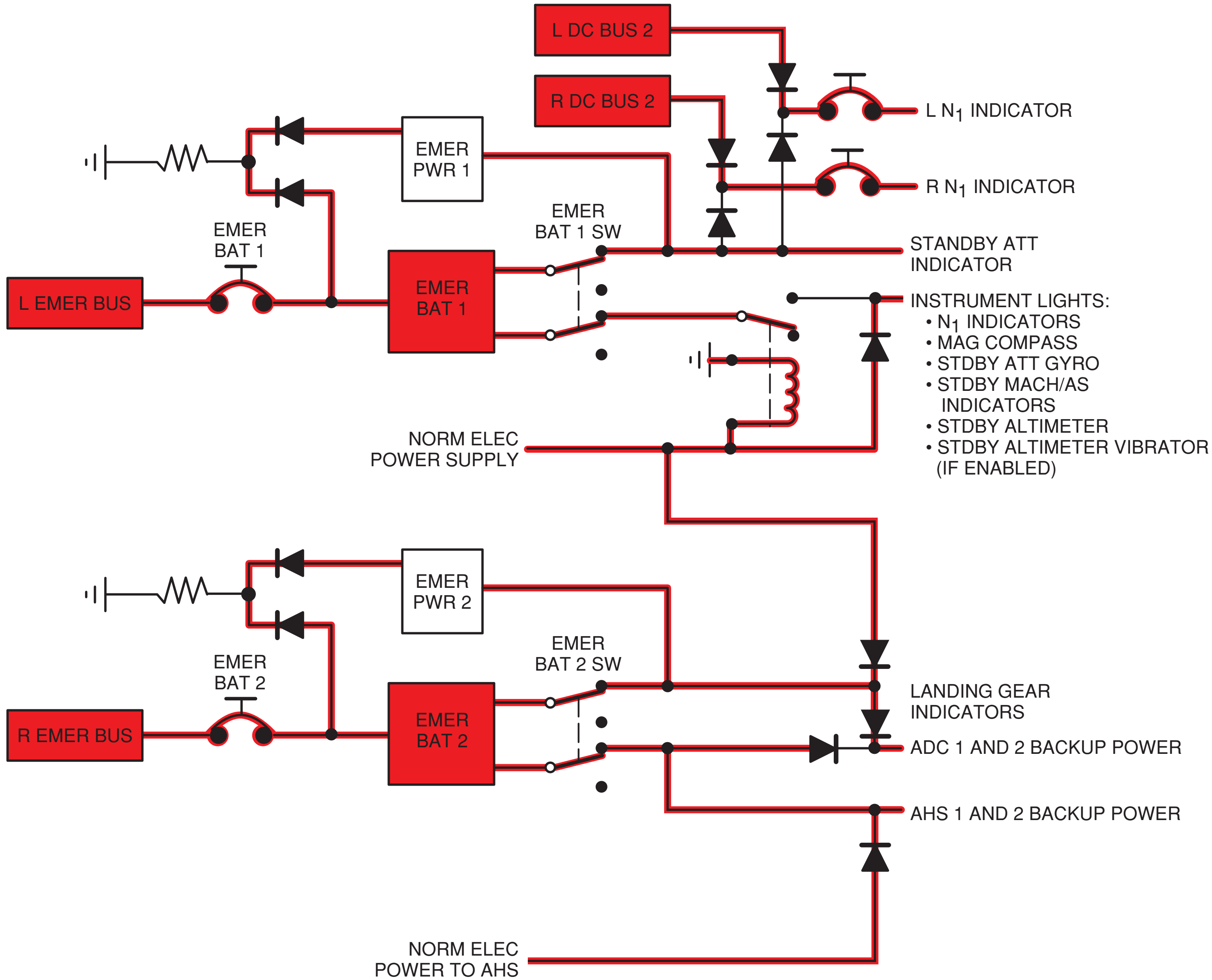


Figure 2-19. Emergency Power Distribution—Normal Aircraft Electrical Power On (Standard Aircraft)

With the emergency batteries ON, the standby attitude indicator (60-001 through 60-248) is always powered from emergency battery No. 1. If power is available from the aircraft electrical system, the emergency battery is constantly being recharged as it provides power for the standby attitude indicator. The other equipment connected to the emergency batteries is normally powered by the aircraft's electrical system. It is powered by the emergency batteries only when all normal electrical power is off or has failed (Figure 2-20).

On modified Learjet 60 aircraft:

- If the normal electrical system has failed and only one emergency backup is installed, EMER BAT 1 power supply will provide electrical power for the standby attitude indicator, N_1 indicators, selected instrument lights (standby attitude indicator, N_1 indicators, standby airspeed indicator, standby altimeter, and magnetic compass), gear position lights, NAV 1 and RTU 1.
- If a second emergency backup battery is installed, EMER BAT 2 will supply electrical power for gear position lights, NAV 1, RTU 1, and backup power for the attitude heading reference system (AHS 1 and 2), air data computers (ADC 1 and 2).
- If a third emergency backup battery is installed, EMER BAT 3 will supply emergency power to FMS 1 and VHF COMM. Additionally, this will allow programming of FMS 1 on the ground when a GPU or APU is not available (Figure 2-21).
- The system is controlled through the EMER BAT 1, EMER BAT 2, and EMER BAT 3 switches on the pilot switch panel. Amber EMR PWR 1, EMR PWR 2, and EMR PWR 3 annunciators on the center instrument panel will illuminate whenever electrical power from the associated emergency power supply is being used.

Operating time of equipment powered by the emergency batteries is presented in the *AFM* (see appropriate supplement).

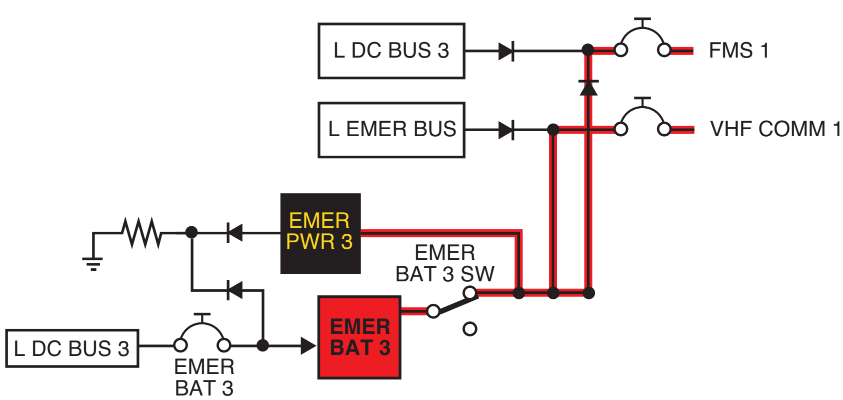


Figure 2-21. Emergency Battery 3 Schematic—Optional

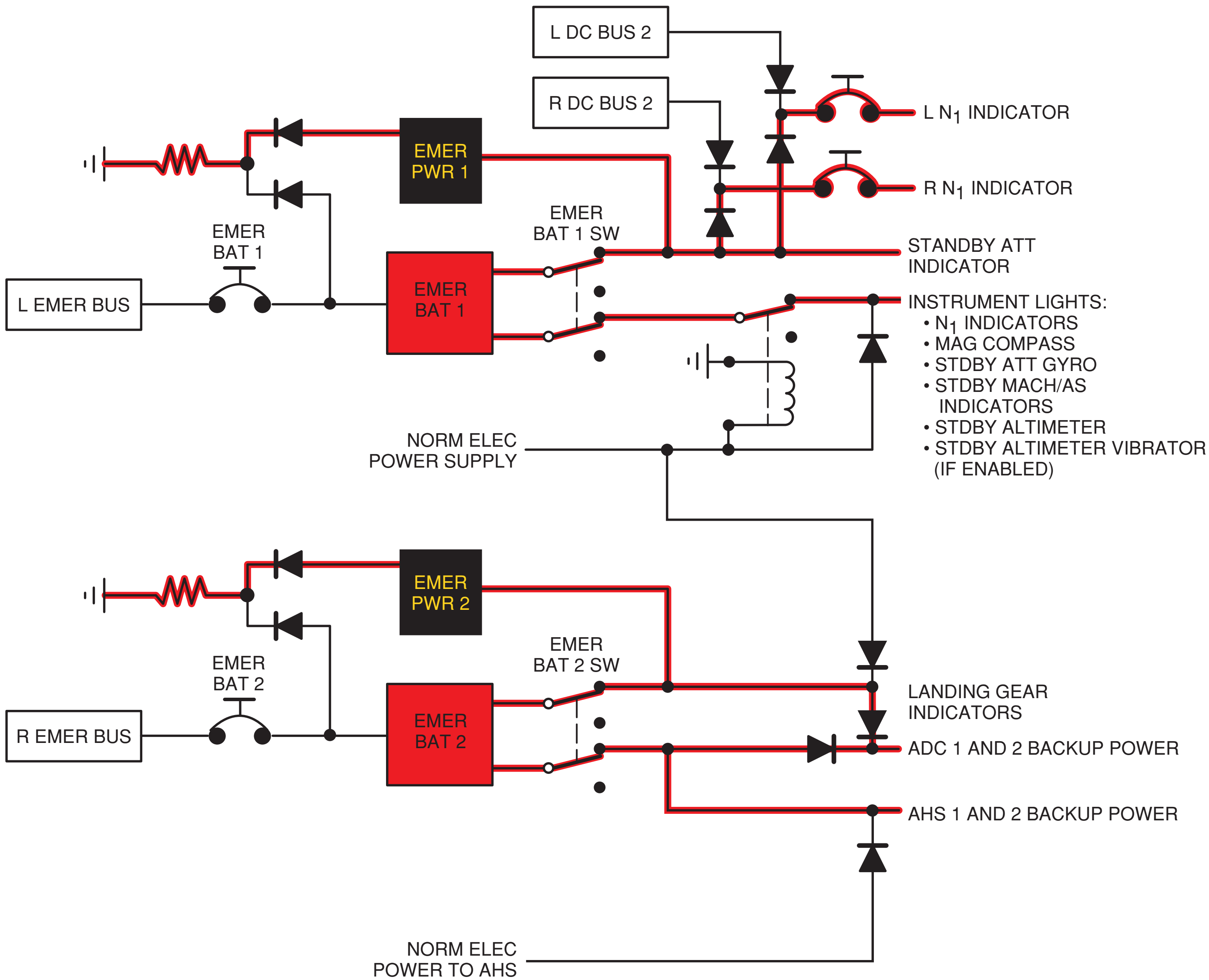


Figure 2-20. Emergency Power Distribution—Aircraft Electrical Power Failed (Standard Aircraft)

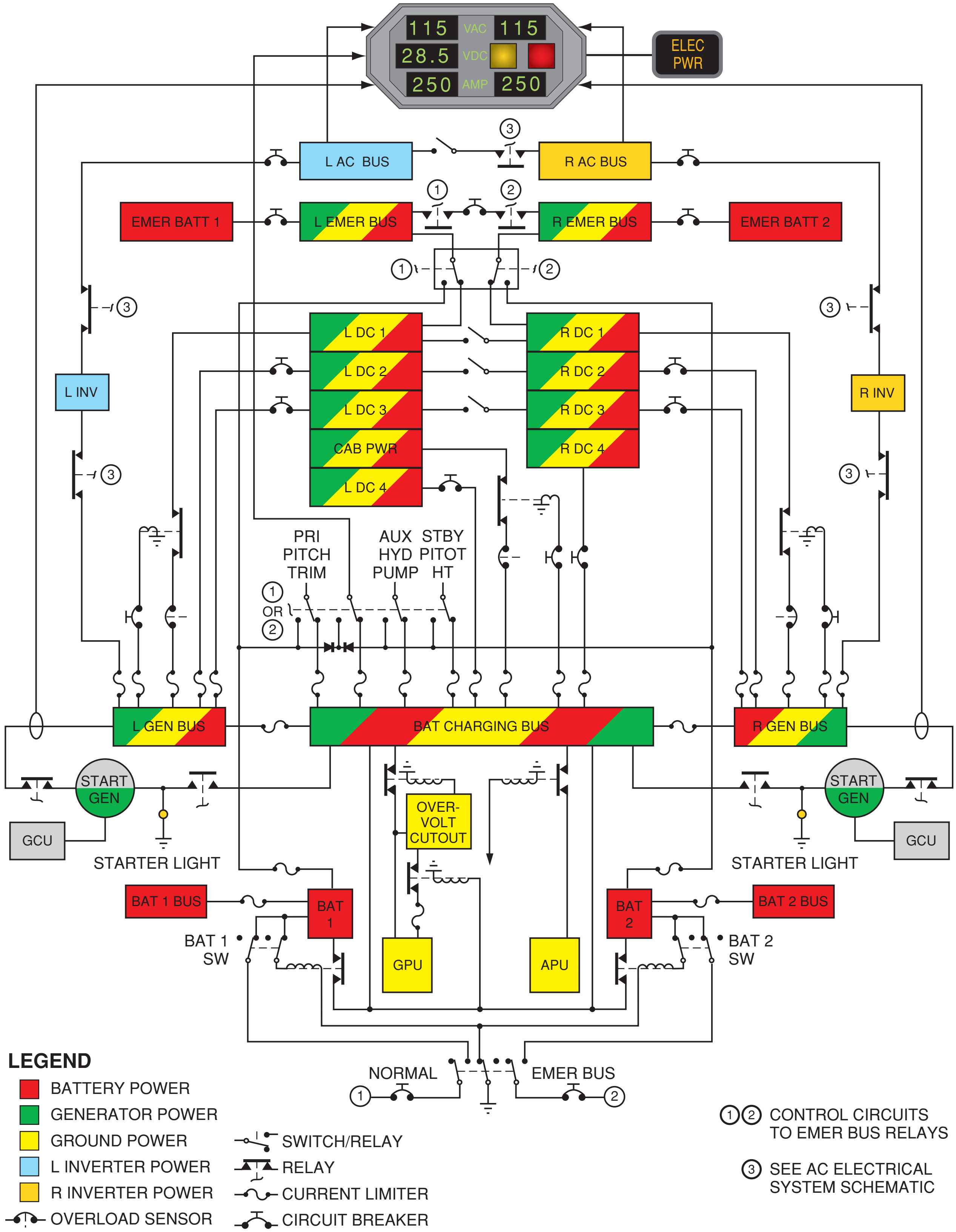


Figure 2-22. Electrical System

QUESTIONS

1. What does the DC voltmeter indicate?
   A. Battery voltage only
   B. Generator voltage only
   C. Voltage on the generator buses
   D. Voltage on the battery charging bus if the EMER BUS–NORMAL switch is in the NORMAL position

2. When a GPU is used for engine start, the output should be what value?
   A. Regulated to 24 volts
   B. Regulated to 28 volts and limited to 1,500 amps maximum
   C. Regulated to 33 + volts
   D. Regulated to 28 volts and limited to 500 amps maximum

3. Which buses can the aircraft batteries power?
   A. Battery buses only
   B. Battery and battery-charging buses only
   C. All buses except the 115 VAC
   D. All buses including the AC through the inverters

4. Which of the following indicates a generator failure?
   A. The electrical power monitor indicates a load.
   B. GEN switch ON, GEN light illuminated, and electrical power monitor indicates no load after completing GEN fail checklist.
   C. GEN light is extinguished.
   D. DC voltmeter reads less than 28 volts.

5. If aircraft electrical power fails, which systems are powered by the emergency batteries?
   A. Standby attitude indicator, emergency DC buses, and emergency inverter
   B. Flap and gear
   C. Standby attitude indicator, N_1 indicators, gear, and flaps
   D. Standby attitude indicator and N_1 indicators; instrument lights for N_1 indicators, mag compass, standby attitude gyro, standby AS, and standby altimeter; landing gear indicators; ADC 1 and 2 and AHS backup power

6. What does illumination of the amber CUR LIM light indicate?
   A. Both 275-amp current limiter are blown.
   B. One or both 275-amp current limiters are blown.
   C. Both 275-amp current limiter are good.
   D. Both DC BUS 3 current limiters are blown.

7. What would failure of the left inverter initially result in?
   A. Loss of power to the left AC bus
   B. Illumination of a red light on the electric power monitor and flashing of the left VAC display
   C. Illumination of the amber ELEC PWR light on the annunciator panel
   D. All of the above

8. Which of the following is true if both 275-amp current limiters fail in flight?
   A. Only aircraft batteries power the L and R DC 4 buses.
   B. Only aircraft batteries power the CAB PWR bus.
   C. The battery-charging bus can only be powered by the aircraft batteries.
   D. All of the above are correct.

9. What does illumination of the red light on the electric power monitor and flashing of the associated VAC display mean?

A. Inverters are operating okay.
B. Inverter output is less than 90 VAC or above 135 VAC on the associated inverter.
C. The associated AC circuit breaker may be open.
D. B or C could be correct.

10. What does illumination of the amber light on the electric power monitor and flashing of the associated amp display indicate?

A. A generator has failed.
B. Amperage draw on the associated generator is between 326–400 amps.
C. The generator control unit has failed.
D. Amperage draw is zero on the associated generator.

11. With aircraft generators failed and the EMER BUS–NORMAL switch placed to the EMER BUS position, which one of the following statements is true?

A. The aircraft batteries will power the battery charging bus plus the DC and AC emergency buses.
B. The aircraft batteries will power the battery buses, DC emergency buses, DC voltmeter, AUX HYD pump, STBY PITOT HT and PRI PITCH TRIM.
C. The aircraft batteries will power all DC and AC equipment that is turned on.
D. The emergency batteries will power the emergency buses.

12. What does an illuminated amber light and an increasing VDC display n the electric power monitor (EPM) indicate?

A. The generator control unit has failed completely.
B. A generator has failed and the other generator is picking up the load.
C. There is a short downstream from one of the generators.
D. One or both of the generators is overvolting.

13. When illuminated, the EMR PWR lights indicate?

A. The emergency batteries have failed.
B. The emergency battery switches are turned off.
C. The emergency battery switches are turned ON and the emergency batteries are not receiving a trickle charge from the aircraft electrical system.
D. The emergency battery switches are turned ON and the EMER BUS–NORMAL switch is in the NORMAL position.

14. Which of the following circuits are powered directly from the battery buses?

A. Emergency battery charging, inverter power, and generator control circuits
B. Emergency directional gyro, emergency gear and flap extension
C. Tailcone inspection light and toilet service receptacle, cabin entry lights, single point pressure refueling and the aft cabin and tailcone baggage compartment lights
D. Left and right stall warning systems.

15. With dual generator failure in flight, how long should fully charged aircraft batteries last with the EMER BUS–NORMAL switch in the EMER BUS position?

A. At least 1 hour
B. 30 minutes
C. 2.5 hours
D. 6 hours

CHAPTER 3
LIGHTING

CONTENTS

ILLUSTRATIONS

CHAPTER 3
LIGHTING

INTRODUCTION

Lighting is used to illuminate the cockpit area and all flight instruments. The majority of the instruments are internally lighted. For general illumination, floodlights, of either the fluorescent or incandescent type, are used. A rheostatically controlled gooseneck map light is installed on both left and right side panels. The standard warning lights are available for the cabin area, and optional emergency lights are available to illuminate the exits in the event of an emergency. Exterior lighting consists of landing-taxi, recognition, strobe, navigation, beacon, and wing inspection lights.

GENERAL

Aircraft lighting is divided into interior and exterior lighting. Interior lighting is further divided into cockpit, cabin (includes lavatory and cabin baggage), tailcone baggage and tailcone maintenance lighting. Cockpit lighting consists of map lights, glareshield floodlights, instrument/indicator lights, panel lights, dome lights and a switch panel to control the lights.

Cabin lighting consists of entry, aisle, overhead, table and read lights, also lavatory and cabin baggage lights.

Tailcone baggage lighting consists of lighting in the tailcone baggage compartment. Tailcone maintenance lighting consists of a light in the tailcone.

Optional emergency lighting in the cabin utilizes the cabin overhead lights and includes additional lights installed at the exits.

INTERIOR LIGHTING

The interior lighting locations for the cockpit, cabin, baggage, and maintenance areas of the aircraft are shown in Figure 3-1.

COCKPIT LIGHTING

See Figure 3-2 and the description in this chapter.

Instrument Panel Floodlights

Three cold-cathode, fluorescent lights are installed under the glareshield to illuminate the instrument panel. The lights are controlled by the FLOOD rheostat switch on the pilot L INSTR LIGHTS panel (Figure 3-3). Power to operate the lights is provided by two 600 VAC power supply units. The power supply units draw 28 VDC through the FLOOD LTS circuit breaker on the pilot LIGHTS group of circuit breakers. The floodlights are also powered when EMER BUS is selected.

NOTE

When initially turning the floodlights on, it is recommended that the rheostat be positioned to full bright for two to three minutes to extend the fluorescent tube operating life.

Instrument Lights

Incandescent lighting is installed for the pilot, copilot instruments, center instrument panel, pedestal indicators, and the magnetic compass. The lights are controlled by the INSTR rheostat switch and CENTER PNL/PEDESTAL rheostat switch on the pilot L INSTR LIGHTS panel and by the INSTR rheostat switch on the co-pilot R INSTR LIGHTS panel. The 28-VDC power for the lights is supplied through the L and R INSTR LTS circuit breakers and the CENTER PANEL-PED LTS circuit breaker on the pilot and copilot LIGHTS group of circuit breakers.

Pilot INSTR Lights

The pilot INSTR dimmer rheostat switch (Figure 3-3) provides lighting control for the engine indicators, L angle of attack (AOA) indicator, left Davtron clock, oxygen press indicator, and electrical power monitor.

NOTE

EMER BAT 1 can also supply power to light the N-1 indicators, mag compass, standby ATT GYRO, standby Mach/ IAS indicator and standby altimeter.

EMER BAT 2 can also supply power to the landing gear indicator during aircraft total electrical power failure.

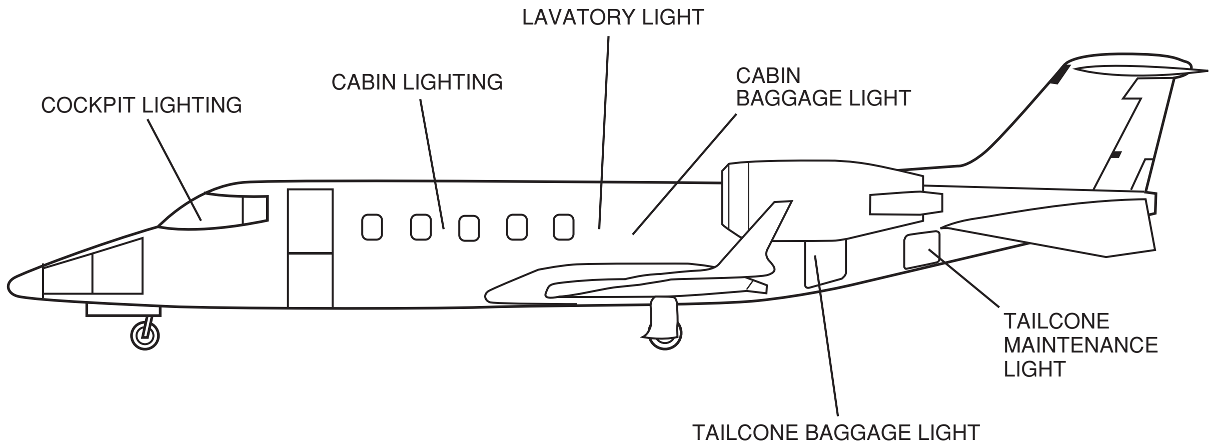
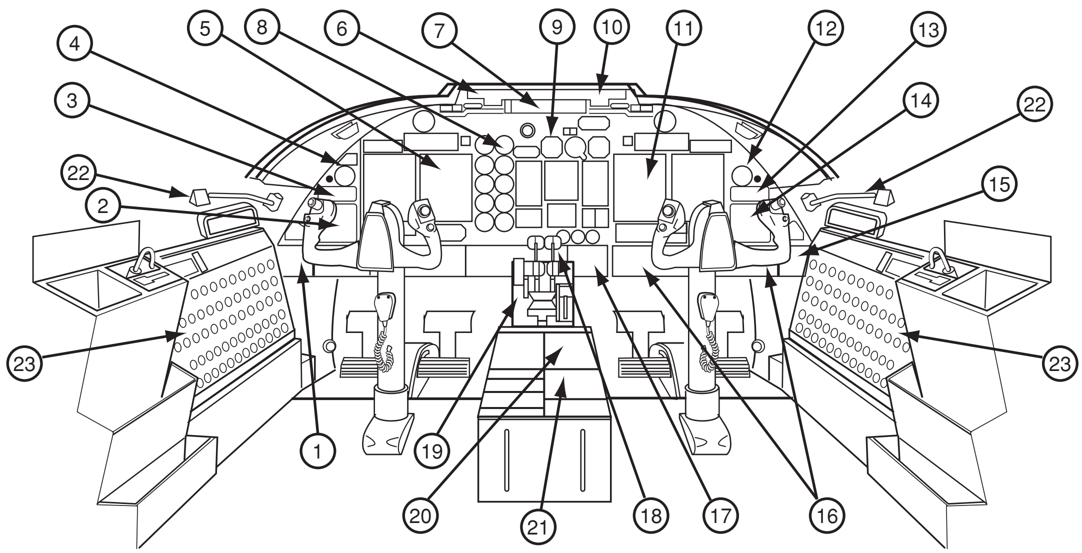


Figure 3-1. Interior Lighting Locations



1. PILOT'S L INSTR LIGHTS SWITCH PANEL
2. PILOT'S AUDIO CONTROL PANEL
3. PILOT'S EFIS CONTROL PANEL (ECP)
4. ANTISKID AND PARKING BRAKE LIGHTS, PILOT'S CLOCK
5. PILOT'S FLIGHT INSTRUMENT PANEL AND GLARESHIELD
6. GLARESHIELD ANNUNCIATOR PANEL
7. AUTOPILOT/FLIGHT DIRECTOR CONTROL PANEL
8. ENGINE INSTRUMENTS, FUEL AND ELECTRICAL INDICATORS
9. CENTER INSTRUMENT PANEL AND RADIO TUNING UNITS
10. GLARESHIELD ANNUNCIATOR PANEL
11. COPILOT'S FLIGHT INSTRUMENT PANEL AND CONTROLS
12. COPILOT'S CLOCK
13. COPILOT'S EFIS CONTROL PANEL (ECP)
14. COPILOT'S AUDIO CONTROL PANEL
15. COPILOT'S R INSTR LIGHTS SWITCH PANEL
16. CABIN CLIMATE AND PRESSURIZATION CONTROL PANEL
17. HYDRAULICS, LANDING GEAR, FLAPS AND SPOILER
18. EXTERIOR LIGHTS SWITCH PANEL
19. THRUST LEVER QUADRANT
20. COURSE HEADING AND CONTROL DISPLAY UNITS
21. TRIM, FUEL, HF RADIO CONTROL PANEL
22. MAP LIGHT L AND R SIDE
23. CIRCUIT BREAKER PANELS

Figure 3-2. Cockpit Lighting

Center PNL/Pedestal

The dimmer rheostat switch on the pilot L INSTR LIGHTS panel controls lighting for: trim indicators (pitch, roll and yaw), standby gyro, standby airspeed and altitude, magnetic compass, course heading/CVR panel, fuel quantity, flap position, spoiler position, wing temperature, hydraulic pressure and air pressure indicators, fuel control panel and trim switch panel. The lights are also powered by the EMER BUS.

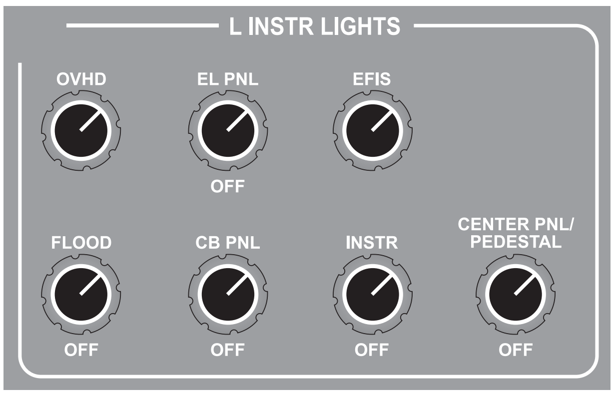


Figure 3-3. L INSTR LIGHTS

Copilot INSTR Lights

The copilot INSTR dimmer rheostat switch provides lighting control for the temperature control indicators, cabin temperature indicator, R angle of attack (AOA) indicator, right Davtron clock, cabin pressure selector and cabin pressure indicator (Figure 3-4).

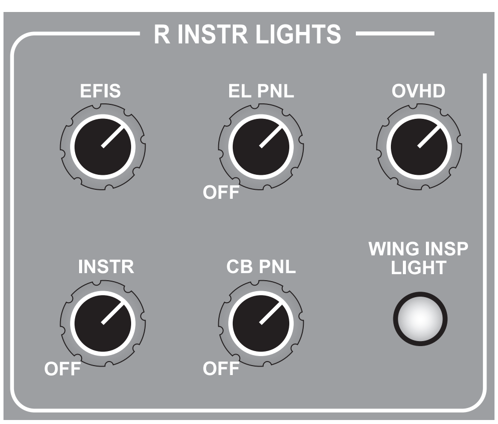


Figure 3-4. R INSTR LIGHTS

Switch Panel Lights, (EL PNL) (CB PNL)

Electroluminescent lighting is used to illuminate switch and circuit-breaker labels on the pilot and copilot switch, and circuit-breaker panels.

Electroluminescent (EL) lighting uses 28-VDC supplied through the L and R EL LTS circuit breakers on the pilot and copilot LIGHTS group of circuit breakers.

EL PNL and the CB PNL rheostat switches are on the pilot and copilot respective L or R INSTR LIGHTS switch panels.

Pilot EL PNL and CB PNL Rheostat Switches

The pilot EL PNL (Figure 3-3) rheostat switch controls lighting of the pilot subpanel, pilot switch panel, center switch panel, land gear panel (does not include unsafe and down lights), test switch panel, anti-skid/park brake annunciator panel and engine sync switch panel. The pilot CB PNL rheostat switch controls the electroluminescent lighting of the pilot circuit-breaker panel.

Copilot EL PNL and CB PNL Rheostat Switches

The copilot EL PNL rheostat switch controls lighting for the copilot subpanel and the copilot switch panel (Figure 3-4). The copilot CB PNL rheostat switch controls electroluminescent lighting of the copilot circuit-breaker panel.

Dome (OVHD) Lights

There are two lights mounted in the cockpit overhead paneling. The OVHD rheostat switch on the pilot L INSTR LIGHTS switch panel controls LH dome light (Figure 3-3).

The OVHD rheostat switch on the copilot R INSTR LIGHTS switch panel controls the RH dome light (Figure 3-4).

These lights use 28-VDC power from the CENTER PANEL-PED LTS circuit breaker on the pilot LIGHT circuit breaker group. Each dome light has a three-position (ON–OFF–REMOTE) rocker switch located adjacent to the light. If the switch(es) is in the ON position (aft selected), the light intensity is controlled by its respective OVHD light rheostat switch on the L or R INSTR LIGHTS switch panel. In this case, the lights are supplied 28-VDC through the CENTER PANEL-PED LTS circuit breaker on the pilot LIGHT group of circuit breakers. If either rocker switch is placed to the REMOTE (forward selected) position, the corresponding dome light will illuminate (full intensity) whenever the ENTRY and COCKPIT DOME LIGHTS switch, located by the cabin entry door (Figure 3-5), is turned on. In this case, aircraft batteries are not required to be on since the ENTRY circuit is powered from the right battery bus through the ENTRY LTS circuit breaker on the CABIN group of circuit breakers. With the switch in the OFF position the dome lights will not illuminate regardless of which switch is used, OVHD or ENTRY and COCKPIT DOME LIGHTS.

EFIS Lights

The pilot EFIS rheostat switch, located on the L INSTR LIGHTS switch panel (Figure 3-3), controls lighting of the pilot PFD (primary flight display), MFD (multifunction display), RTU (radio tuning unit), SDU (sensor display unit), and ECP (EFIS control panel). The NAV light switch will also dim the ECP (EFIS control panel) as long as the EFIS rheostat switch is not in the OFF position.

NOTE

Each PFD, MFD, RTU, SDU also has an individual dimmer switch located on the upper left or right side of the instrument.

Electrical power is supplied through the EFIS CTL-1 circuit breaker on the pilot INSTRUMENTS group of circuit breakers.

The copilot EFIS rheostat switch, located on the R INSTR LIGHTS switch panel (Figure 3-4), controls lighting of the copilot PFD (primary flight display) MFD (multifunction display), RTU (radio tuning unit), and ECP (EFIS control panel). The NAV lights switch will also dim the ECP (EFIS control panel) as long as the EFIS rheostat switch is not in the OFF position.

NOTE

Electrical power is supplied through the EFIS CTL-2 circuit breaker on the copilot INSTRUMENTS group of the circuit breakers.

Map Reading Lights

Map reading lights are located on the left and right cockpit sidewalls (Figure 3-2). Each light is attached to the end of the flexible conduit (gooseneck) and is controlled by a rheostat located on the top of the mounting assembly. The lights use DC power, supplied through the L or R INSTR LTS circuit breaker located on the pilot or copilot LIGHTS group of circuit breakers.

NOTE

If left in the ON position, the map reading light and its cover can become very hot.

CABIN LIGHTING

See Figures 3-1 and 3-5 and the descriptions in this section.

General

Passenger compartment lighting consists of entry, aisle, overhead, baggage, passenger read/table, lavatory (read/vanity), refreshment cabinet, and NO SMOKING/FASTEN SEAT BELT lights. Additional lights are provided in the tailcone baggage compartment and the tailcone maintenance area.

Entry Light

The entry light system consists of a ENTRY and COCKPIT DOME LIGHTS switch located near the entry door (Figure 3-5), the light located just above the entry steps and an ENTRY LTS circuit breaker located on the copilot CABIN group of circuit breakers. The light is hot wired to the right battery bus and is operable regardless of battery switch position. This switch also controls the cockpit dome lights when their switches are selected to REMOTE.

Aisle Lights

The aisle lights consist of lights installed on each side of the center aisle to provide foot path lighting. The lights are controlled by the AISLE LIGHTS switch, located near the entry door (Figure 3-5). Electrical power is supplied through the AISLE LTS circuit breaker on the pilot CABIN group of circuit breakers.

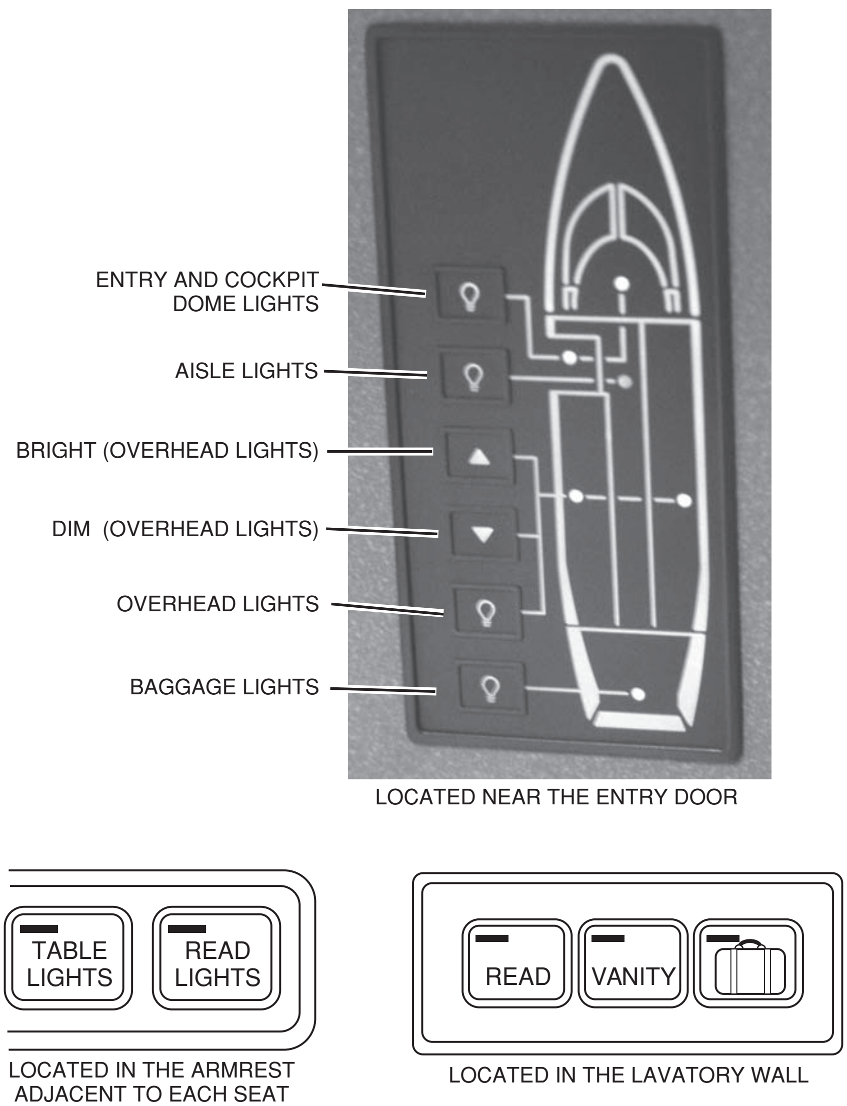


Figure 3-5. Cabin Lighting Controls (Typical)

Overhead Lights

The cabin overhead lighting is provided by cold-cathode, fluorescent lighting recessed in the sides of the cabin center headliner.

Control of the lights is by three switches on the lights panel near the entry door (Figure 3-5). The switches are labeled BRIGHT, DIM and OVERHEAD LIGHTS. One switch provides the on/off function, while the other two provide bright and dim functions. Electrical power is supplied through the CABIN LTS circuit breaker on the pilot CABIN group of circuit breakers. These lights are hot wired to the right battery bus and are operable regardless of battery switch position.

NOTE

To avoid damage to the fluorescent tubes or the inverter units, the lights should be operated full bright for 2 to 3 minutes before selecting dim.

During engine start, the lights should be turned off or operated at full bright. In the event of cabin depressurization, the lights will automatically illuminate full bright when the cabin altitude reaches approximately 14,500 feet. On aircraft equipped with the optional emergency lighting system, the overhead lights illuminate automatically in the event of aircraft electrical system failure. When the lights are illuminated by the emergency system, they receive power from three emergency batteries.

Baggage Compartment Lights

Cabin baggage compartment lighting (Figure 3-6) consists of two overhead lights. The lights are controlled by a BAGGAGE LIGHTS switch, located near the entry door (Figure 3-5) or through a BAGGAGE light switch in the lavatory. The light's circuits are hot-wired to the right battery bus through the AFT BAG LT DC circuit breaker on the copilot CABIN group of circuit breakers. Therefore, the lights are operable regardless of aircraft BATTERY switch position.

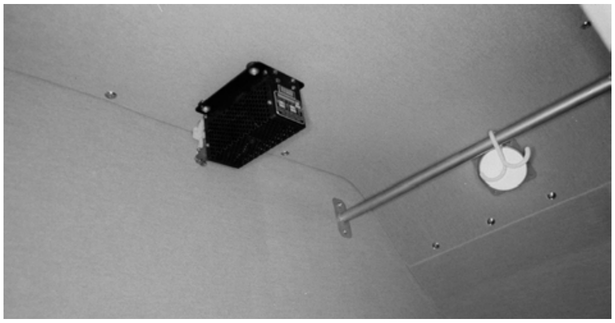


Figure 3-6. Baggage Compartment Lights

Passenger Reading Lights

The passenger reading lights (Figure 3-7) are installed in the convenience panels above the seats on each side of the cabin. Some convenience panels consist of an eyeball-type air outlet and a reading light, while others consist of a two-light assembly, referred to as table lights. Each light includes an integral, directionally-adjustable lens.

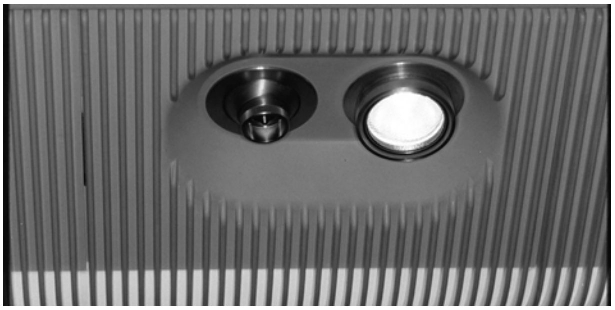


Figure 3-7. Cabin Reading Lights

These lights are controlled by the READ LIGHTS and TABLE LIGHTS switches (Figure 3-5), located in the armrest adjacent to each seat location. DC circuit power is supplied through the LH READ LTS and RH READ LTS circuit breakers on the pilot CABIN group of circuit breakers.

Lavatory Lights

Lavatory lighting (Figure 3-8) consists of cold-cathode fluorescent lighting, recessed in the lavatory headliner, a reading light in the RH overhead convenience panel, and a vanity light, installed on the LH side, over the vanity light cabinet. The cold-cathode fluorescent lighting is controlled along with the other passenger overhead fluorescent lighting (see Overhead Lighting). The VANITY light (Figure 3-5) is controlled by a switch located on the lavatory wall. READ light circuit power is supplied through the RH READ LTS circuit breaker on the pilot CABIN group of circuit breakers; the VANITY light circuit breaker is located on the pilot CABIN group of circuit breakers.

Figure 3-8. Lavatory Lights

Cabinet Lights

The cabinet light system consists of various lights within the refreshment cabinet, micro-switches actuated by doors or drawers. Circuit power for the cabinet lights is supplied through the CABINET LTS circuit breaker on the pilot CABIN group of circuit breakers.

NOTE

All light switch panels in the cabin are backlighted. Power for backlighting is supplied through the switching module through the LH READ LTS circuit breaker.

Passenger Warning Lights

The NO SMOKING/FASTEN SEAT BELTS and FASTEN SEAT BELTS warning light system consists of two sets of warning lights (Figure 3-9), one switch on the center switch panel, and a PASS INFO circuit breaker on the copilot CABIN group of circuit breakers. The switch has three positions (NO SMOKING/FASTEN SEAT BELTS–OFF–FASTEN SEAT BELTS). When the switch is moved from OFF to either position, an audible chime sounds and the appropriate symbols illuminate. Additionally, a RETURN TO SEAT sign is installed in the lavatory. The RETURN TO SEAT sign will be illuminated whenever the fasten seat belt sign is illuminated.

Figure 3-9. Passenger Warning Lights

The chime is generated by the passenger speaker amplifier, and broadcast through the passenger speakers.

Tailcone Baggage Lights

Two lights are installed in the tailcone baggage compartment to provide illumination of the compartment. The lights are controlled, along with the cabin baggage compartment lights, by the BAGGAGE LIGHTS switch near the entry door.

3 LIGHTING

Tailcone Maintenance Lights

The tailcone maintenance lighting consists of a tailcone light assembly, a MAINT LIGHTS-OFF switch (Figure 3-10), and a door actuated switch, installed on the lower edge of the door opening. The toggle switch and door actuated switch are wired in series to the light assembly; therefore, the tailcone access door must be open and the toggle switch set to the MAINT LIGHTS position to illuminate the light. When the toggle switch is set to OFF, the light will extinguish regardless of the access door position. When the access door is closed, the light will extinguish regardless of toggle switch position. Circuit power is provided from the No. 1 battery through a 5-amp current limiter.

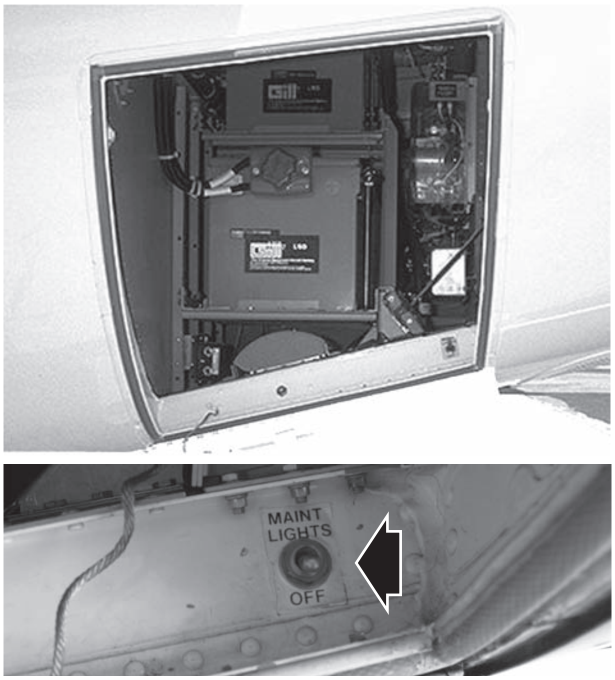


Figure 3-10. Tailcone Maintenance Lights

EMERGENCY LIGHTING SYSTEM (OPTIONAL)

The emergency lighting system provides cabin and exit lighting in the event of a failure of the normal electrical system. The system consists of two emergency batteries, an upper cabin entry door light, an emergency exit/baggage door light, the cabin overhead fluorescent lights, two system switches, and associated aircraft wiring. The batteries are charged through the 5-amp EMER LTS circuit breaker, on the copilot LIGHTS group of circuit breakers. If armed, the system will automaticly activate whenever R DC BUS 4 loses normal electrical power. The system will automatically activate during EMER BUS mode of operation.

EMER LIGHT Switch

The EMER LIGHT switch, located on the center switch panel (Figure 3-11), provides the test function for the system and for automatic illumination of the emergency lights in the event of an interruption of normal electrical power (RDC BUS 4). The switch has three positions (TEST, ARM, and DISARM).

Setting the switch to TEST simulates a failure of the normal electrical power system and illuminates the upper cabin entry door lights, the emergency exit/baggage door light, and the cabin overhead fluorescent light. Setting the switch to ARM will arm the system to illuminate the emergency lights in the event of a failure of the normal electrical system. Setting the switch to DISARM isolates the emergency lights from the emergency batteries. The switch should be set to ARM prior to takeoff. If the switch is in the DISARM position and at least one BAT switch is on, the amber light adjacent to the switch will illuminate to remind the pilot that the switch should be set to ARM. The switch should be set to DISARM prior to setting the BAT switches to OFF.

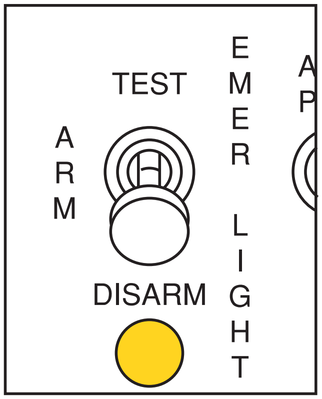


Figure 3-11. Emergency Lighting and Control

EMER LTS-NORM Switch (Optional)

The EMER LTS-NORM switch, located on the left service cabinet near the entry door, provides for manual illumination of the emergency lights. When the switch is set to EMER LTS, the upper cabin entry door light, the emergency exit/baggage door light, and the cabin overhead fluorescent lights will illuminate.

For normal operation, the switch should be set to NORM. Setting the switch to NORM does not hinder automatic illumination of the emergency lights in the event of a failure of the normal electrical system.

EXTERIOR LIGHTING

The exterior lighting systems consist of the landing-taxi lights, a recognition light, navigation lights, tail logo lights (optional), anticollision (beacon/strobe) lights, wing inspection light and (optional) exterior convenience lights (Figure 3-12).

LANDING-TAXI LIGHTS

The landing-taxi lights system consists of one lamp mounted on each landing gear strut, one 20-amp fuse for each side in the current limiter panel, relays and dimming resistors, and L and R LDG LT switches in the LIGHTS group of switches on the center switch panel. The landing light switches have three positions: OFF, TAXI, and LDG LT L or R (Figure 3-13).

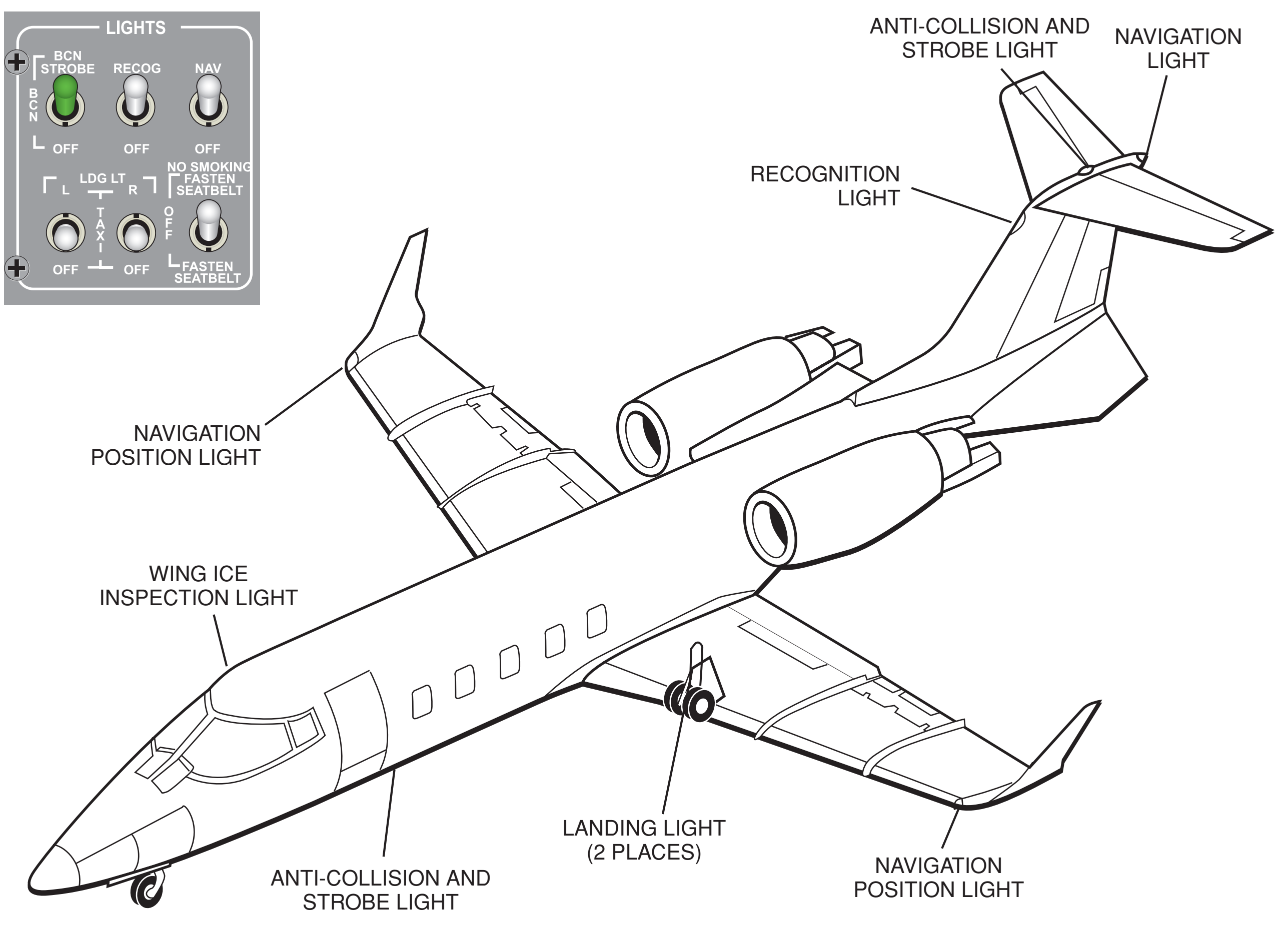


Figure 3-12. Exterior Lighting Locations and Switch Panel

Figure 3-13. Landing Lights and Control

Setting the L and R LDG LT to LDG LT causes relays to close sending 28 VDC from the L and R GEN buses to the landing lights. Setting the switches to TAXI causes relays to close, sending 21 VDC through resistors to dim the lights. Regardless of switch position, the lights will not illuminate unless the respective landing gear down and locked switches are closed to provide a ground.

NOTE

It is recommended that the lights be operated in the LDG LT mode as sparingly as possible. Bulb service life is shortened approximately 30 times faster in the LDG LT mode. Bulb service life is much longer in the TAXI mode.

Some aircraft are equipped with a pulsating landing light option. On these aircraft, a pulse controller unit controls the landing lights by delivering pulsating DC current at approximately 45 cycles per minute. The effect of this pulsating current is to cause the bulb's brightness to continually vary between approximately 40 percent and 100 percent of full bright. This feature results in greater bulb life. This pulsating feature is activated when the landing gear are down and locked, the RECOG light switch is set to the PULSE position, and the L and R LDG LT switches are set to the OFF position. The pulse feature will not function with the switches in the L or R LDG LT or TAXI positions.

RECOGNITION LIGHT

The recognition light system consists of a light installed in the upper forward leading edge of the vertical stabilizer (Figure 3-14), a RECOG light switch in the LIGHT group of switches on the lower center switch panel, a 20-amp fuse in the current limiter box and a control relay. An optional pulse controller unit with a PULSE RECOG LT circuit breaker on the copilot LIGHTS circuit breaker group may also be installed.

The RECOG switch, when placed to RECOG, closes a relay that applies DC current from the battery charging bus to power the recognition light.

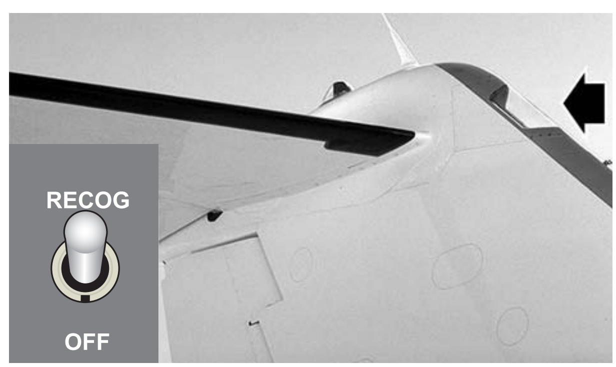


Figure 3-14. Recognition Light and Control

NOTE

For greater lamp life, it is recommended that the recognition light be turned OFF at altitudes of 18,000 feet or above.

Some aircraft are equipped with a pulsating recognition light option. On the aircraft, the RECOG light switch has a third position labeled PULSE and a pulse controller unit. When the switch is placed in the PULSE position, DC current from the PULSE RECOG LT circuit breaker is applied to the pulse controller unit which in turn lights the recognition light by delivering pulsating DC current at approximately 45 cycles per minute. The effect of this pulsating current is to cause the bulb's brightness to continually vary between 40 percent and 100 percent of full bright.

This feature results in enhanced aircraft recognition and longer bulb life.

NAVIGATION LIGHTS

The navigation lights system (Figure 3-15) consists of lights in the outboard leading edge of each wing, and in the upper aft fairing of the vertical stabilizer, a NAV light switch, and DC NAV LTS circuit breaker on the pilot LIGHTS group of circuit breakers.

All three navigation lights are controlled by the NAV light switch. Additionally, setting the NAV light switch to NAV automatically dims the following lights:

- Auto pilot controller
- PARK BRAKE lights
- ANTISKID lights
- START lights
- IGNITION lights
- Pressurization fault/man light

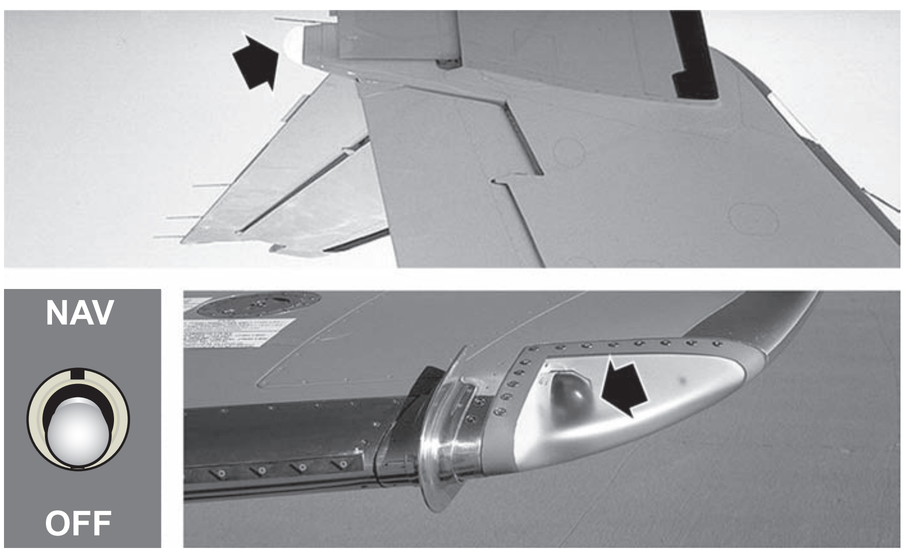


Figure 3-15. Navigation Lights and Control

- Fuel control panel lights
- CVR TEST light
- Pressurization EMER DEPRESS light
- Landing gear UNSAFE/DOWN lights
- SELCAL panel lights
- EFIS reversionary mode lights

Additionally, any radios selected (push/pop rotary knob out) on the communications control panels will light up when the NAV position is selected.

TAIL LOGO LIGHTS (OPTIONAL)

Optional tail logo lights consist of lights installed in the horizontal stabilizer, on either side of the vertical stabilizer, a NAV switch in the LIGHTS group on the center switch panel and a DC LOGO LT circuit breaker on the copilot LIGHTS group of circuit breakers.

When the NAV lights switch on the LIGHTS panel is placed to NAV the tail logo lights will illuminate, lighting both sides of the vertical stabilizer in addition to turning on the navigation lights. Some aircraft have a three-position switch, NAV LOGO–NAV–OFF.

ANTICOLLISION (BEACON/STROBE) LIGHTS

Anticollision (beacon/strobe) lights (Figure 3-16) consist of lights mounted on top of the vertical stabilizer and on the bottom of the fuselage, the BCN/STROBE–BCN–OFF switch on the LIGHTS switch panel, and a BEACON–STROBE LTS circuit breaker on the copilot LIGHTS group of circuit breakers.

Each light incorporates two flash tubes, one with an aviation red filter and one with a clear filter. The lights are controlled through the BCN/STROBE lights switch in the LIGHTS group on the center switch panel. When the switch is placed in the BCN/STROBE position, the red flashtube in each light will flash if the aircraft is on the ground (squat switches ground mode), or the clear flashtube will flash if the aircraft is airborne (squat switch air mode). When the switch is placed in the BCN position, the red flashtube in each light will flash whether the aircraft is on the ground or airborne. Therefore, when the clear strobe is not desired in flight, the switch must be set to BCN or OFF. Each flashtube pulses at a rate of approximately 50 pulses per minute.

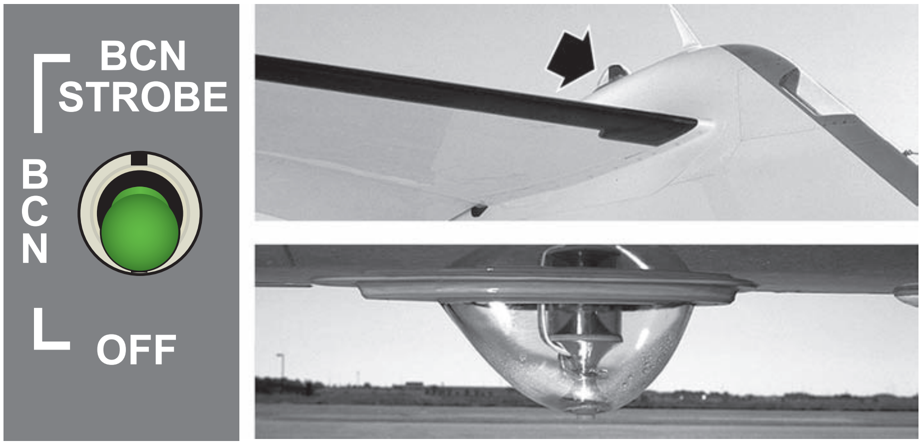


Figure 3-16. Beacon/Strobe Lights and Control

WING INSPECTION LIGHT

The wing inspection light system consists of a WING INSP LIGHT button (Figure 3-17) on the copilot R INSTR LIGHTS switch panel, a light assembly in the RH fuselage skin just below the copilot side window, and a DC WING INSP LT circuit breaker on the copilot LIGHTS group of circuit breakers.

Power is applied to the wing inspection breaker, and the system ground is provided by depressing the WING INSP LIGHT button. This is not a latching type switch, the light will only remain on as long as the button is depressed. When the light is on, it illuminates a spot on the outboard of the RH wing leading edge (Figure 3-17). The black spot enhances visual detection of ice accumulation; however, clear ice may not be detectable by visual inspection.

EXTERIOR CONVENIENCE LIGHTS (OPTIONAL)

The exterior convenience lighting option when installed consists of a light on the underside of each engine pylon. The lights illuminate the area around the tailcone baggage compartment and the single-point pressure refueling access. The lights are controlled by the entry light switch located near the entry door. These lights are inoperative inflight.

FLASH LIGHTS (OPTIONAL)

An optional FLASH LTS circuit breaker, located on the copilot LIGHTS group, provides an in-aircraft method of recharging personal flashlights. Location of the recharging receptacle is an owner preference item and location may vary from aircraft to aircraft.

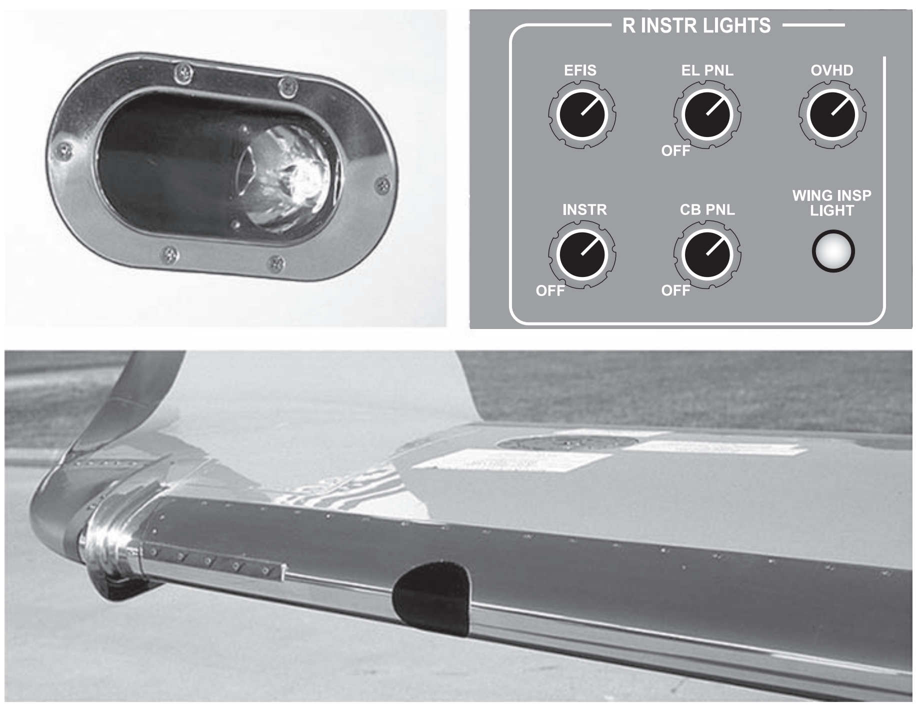


Figure 3-17. Wing Inspection Lights and Control

QUESTIONS

1. Where is the instrument panel floodlight control?

A On the light
B. Just forward of the warning panels
C. On the pilot L INSTR LIGHTS panel
D. On the copilot R INSTR LIGHTS panel

2. In order to extend the service life of the landing-taxi light bulbs:

A. Use the LDG LT position as much as possible
B. Use the LDG LT position as sparingly as possible
C. Never use the landing lights during daylight
D. Use the TAXI position as sparingly as possible

3. Where is the aisle light switch located?

A. RH forward refreshment pedestal
B. In the cockpit
C. Inside the entry door on the left
D. On the light

4. When the cabin OVERHEAD LIGHT switch is turned on, which position should be selected first?

A. ON
B. OFF
C. DIM
D. BRIGHT

5. Which lights come on automatically as the cabin altitude increases above 14,500 feet?

A. Instrument panel floodlights
B. Cabin overhead lights
C. Navigation lights
D. Strobe lights

6. During normal operation, which position should the emergency lighting switch be in?

A. DISARM
B. ARM
C. TEST
D. EMER LT

7. With emergency light installed and the emergency lighting switch in ARM, which lights will come on with aircraft electrical power failure?

A. Aisle lights
B. Lavatory lights
C. Cabin overhead panel lights
D. Reading lights

8. Where is the wing inspection light switch?

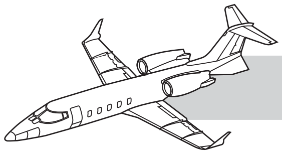
A. On the pilot L INSTR LIGHTS panel
B. On the light
C. On the overhead panel
D. On the copilot R INSTR LIGHTS panel

9. Which of the following require an aircraft inverter to be operating?

A. CB PNL and EL PNL
B. FLOOD
C. INSTR
D. NONE

10. Which of the following lights can be operated with the aircraft batteries turned OFF?

A. Entry lights
B. Baggage lights
C. Aisle lights
D. A and B are correct



CHAPTER 4
MASTER WARNING SYSTEM

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ILLUSTRATIONS

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CHAPTER 4
MASTER WARNING SYSTEM

INTRODUCTION

The master warning system provides a warning for aircraft equipment malfunctions, an unsafe operating condition requiring attention, an abnormal system malfunction, or an indication that a system is in operation.

GENERAL

The main annunciator panel is located on the center portion of the glareshield above and on either side of the autopilot/flight director control panel. In addition to the main annunciator panel, there are also warning, caution, and advisory lights located in various other positions on the instrument panel and on the center pedestal.

A number of aircraft systems include aural warnings for abnormal conditions or malfunctions. These aural warning systems are covered in the chapters that address the related system. A takeoff warning system (horn) not exclusively related to any particular system is installed; therefore, the takeoff warning system is covered in this chapter.

TAKEOFF WARNING SYSTEM

The takeoff warning system (Figure 4-1) activates a warning horn whenever the aircraft is on the ground and the right thrust lever is above 82% TLA (Thrust Lever Angle) if any of the following conditions exist:

- Flaps not set for takeoff
- Spoilers not retracted
- Pitch trim not in a safe condition for takeoff (T.O. TRIM annunciator illuminated)
- One or both thrust reversers unlocked or deployed
- Park brake not released

When the horn is activated, it can be silenced by retarding the thrust levers below 82% TLA or by ensuring that all of the parameters listed above are met. The squat switch relay box inhibits the warning horn while airborne.

NOTE

If the Parking Light bulb is burned out, the warning horn will sound even if the parking brake T handle is full forward.

MAIN ANNUNCIATOR PANEL

There are four rows of red, amber, white, and green lights which are used to announce various malfunctions, failures, and switch positions (Figure 4-2). See also, Figure ANN-1 in the Annunciator Panel section of this manual. Generally, red lights are used to warn of a hazard which requires corrective actions. Amber lights denote cautionary items. White lights denote a system condition that is not normal, but is not critical or cautionary. Green lights indicate conditions which may be normal but need to be monitored by the crew (see Annunciator Panel section). If a glareshield light illuminates and the condition is corrected, the light extinguishes; but if the condition recurs, the light illuminates again. If a white light illuminates in flight, depressing the master warning/caution switch will cause the white lights to go out. Ten seconds after landing, the white lights will come on again as a maintenance advisory (Figure 4-2). White lights can be reactivated in flight by depressing the annunciator test switch.

NOTE

On the ground the white SPOILER EXT light is the only white light that will cancel if the Master Caution is depressed. In the air, the white ENG CMPTR will not cancel if the on-side amber ENG CMPTR light is also illuminated.

MASTER WARNING/ CAUTION LIGHTS

When a red glareshield light illuminates, the red portion of the master WARN/CAUT lights (see Annunciator Panel section) on the pilot and copilot instrument panels flash. Depressing either master WARN/CAUT light causes both pilot and copilot master WARN/CAUT lights to extinguish.

When an amber glareshield light illuminates, the amber portion of the master WARN/CAUT lights (see Annunciator Panel section) on the pilot and copilot instrument panels flash. Depressing either master WARN/CAUT light causes both pilot and copilot master WARN/CAUT lights to extinguish. The master caution (amber light) may be inhibited on the ground by depressing and holding (approximately three seconds) either master caution switch until the master caution lights illuminate. Depressing the annunciator test switch will cancel the master caution (amber light) inhibit. Airborne, the master caution feature will revert to the normal (uninhibited) mode.

The applicable red or amber glareshield light remains illuminated until the malfunction is cleared.

TEST

The glareshield lights are tested by pressing either of the two TEST buttons (see Annunciator Panel section) on the front of the glareshield just outboard of the ENG FIRE PULL handles. The test illuminates the annunciator lights on the glareshield, the pilot and copilot instrument panels, the pilot switch panel, the center switch panel, and the center pedestal. The left and right fuel scavenge pumps are energized during the annunciator test. Scavenge pump operation can be verified by audible indication. Annunciators not tested with the test button are the EMER PWR 1 & 2 annunciators for the emergency batteries. When the test button is released the CABIN FIRE annunciator will remain illuminated for up to 30 seconds while the smoke detector performs a self-test (STEADY light).

DIM

The photo electric cells, next to the TEST buttons, automatically adjust the glareshield annunciator light intensity for existing cockpit light indications. Additional instrument panel and pedestal annunciator lights dim when the NAV light switch is turned on. The remaining lights are controlled with dimmer rheostats.

POWER

Electrical power for the glareshield annunciator panel is supplied from both the left and right WARN LTS circuit breaker on the pilot/copilot Lights circuit breaker group. Either circuit breaker can provide the power to illuminate the annunciators. Both WARN LTS circuit breakers receive power from the left and right EMER buses, respectively.

BULB CHANGE

Removal of annunciator bulbs require maintenance action and special tools.

ILLUMINATION

From the top and left to right, Table 4-1, on the following pages, shows the annunciator panel light label, color, condition for illumination, pilot action, and possible equipment loss. Other listed lights are either on the glareshield, instrument panel, center pedestal, or are optional.

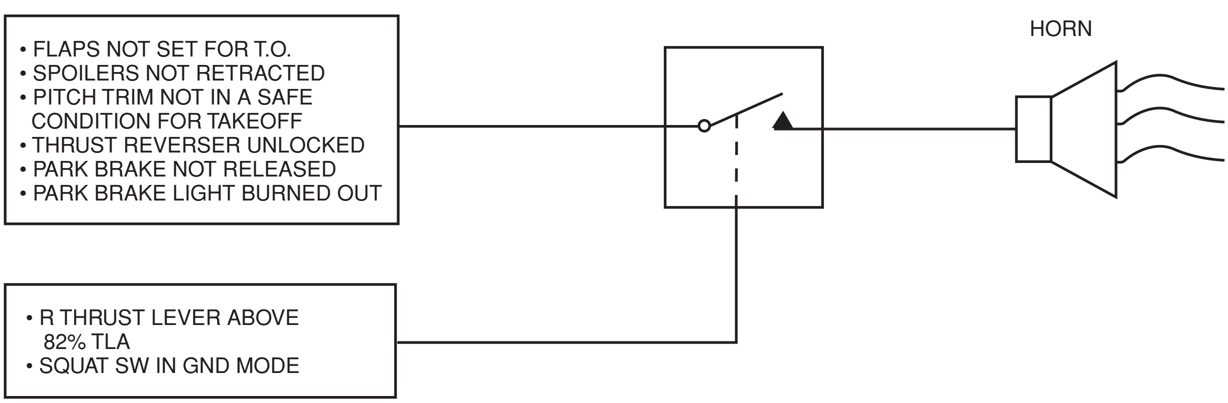


Figure 4-1. Takeoff Warning System

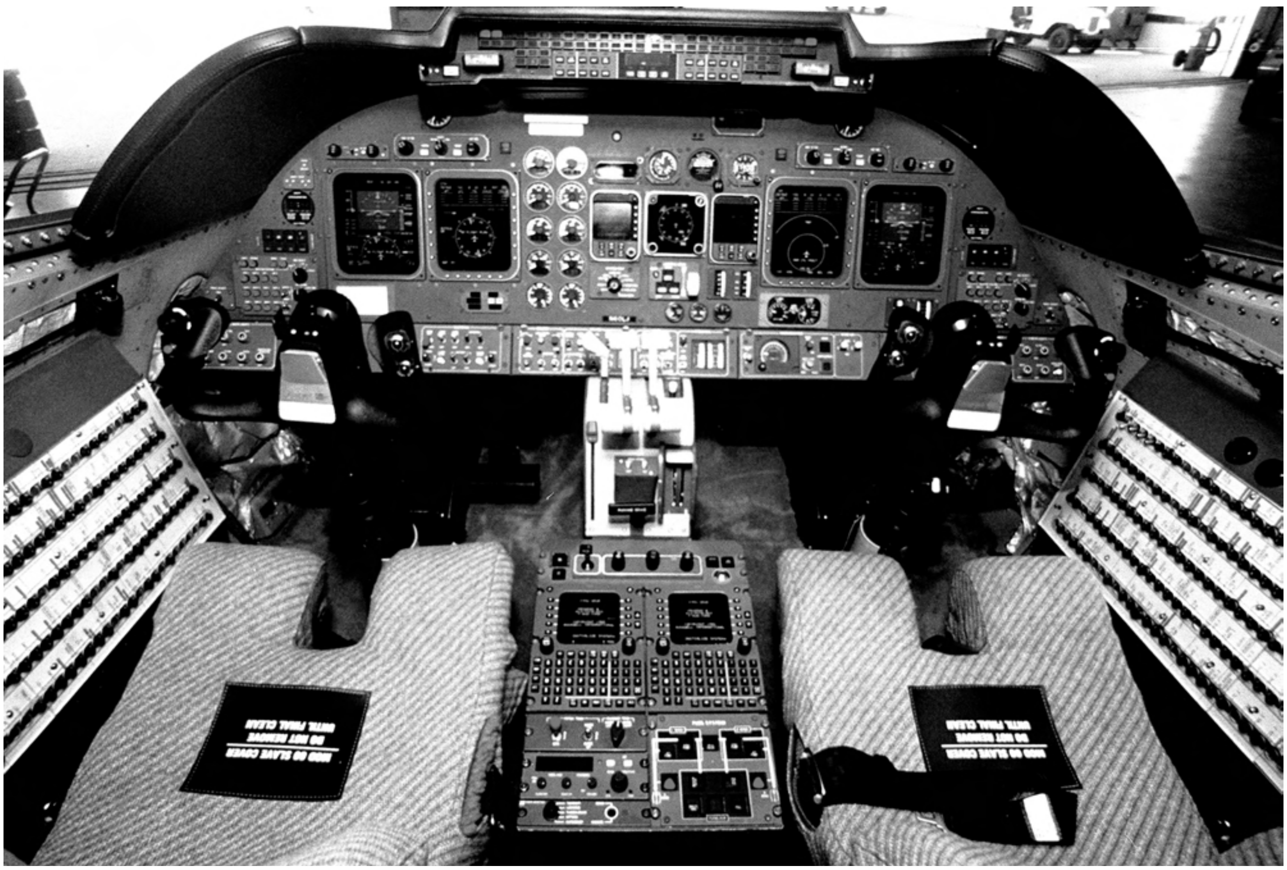


Figure 4-2. Cockpit

If warning, caution, or advisory lights are installed in the cockpit, they must, unless otherwise approved by the Administrator, be (a) Red, for warning lights (lights indicating a hazard which may require immediate corrective action); (b) Amber, for caution lights (lights indicating the possible need for future corrective action); (c) Green, for safe operation lights; (d) Any other color, including white, for lights not described in paragraphs (a) through (c) of this section, provided the color differs sufficiently from the colors prescribed in paragraphs (a) through (c) of this section to avoid possible confusion.

It is suggested that you fold out Figure ANN-1 in the Annunciator Panel section while you review the following annunciators. This will assist you in learning the relative location of each annunciator.

Table 4-1. ANNUNCIATORS

WARNING/ CAUTION/ ADVISORY	PROBABLE CAUSE	PILOT ACTION
L OIL PRESS	Left engine oil pressure is below 20 psi.	Refer to OIL PRESS Warning Light Illuminated in the “Emergency Procedures” section of the *AFM*.
L FUEL PRESS	Less than 2.75 psi fuel pressure to engine-driven pump. Light extinguishes at 3.75 psi.	Check that the jet pump switch is in the ON position and refer to FUEL PRESS Warning Light Illuminated in the “Emergency Procedures” section of the *AFM*.
L ENG CHIP	Metal particles are detected in engine oil.	Refer to ENG CHIP Light Illuminated in “Abnormal Procedures” section in the *AFM*.
L ENG VIB	Engine vibration levels have exceeded limits.	1. Monitor Engine Instruments 2. Refer to Engine Vibration Light Illuminated in the “Abnormal Procedures” section of the *AFM*.
L HYDR PRESS	A low-pressure condition exists on the left engine hydraulic pump (below 150 psi).	Refer to HYDR PRESS Light(s) Illuminated in the “Abnormal Procedures” section of the *AFM*.
R HYDR PRESS	A low-pressure condition exists on the right engine hydraulic pump (below 150 psi).	Refer to HYDR PRESS Light(s) Illuminated in the “Abnormal Procedures” section of the *AFM*.
SPOILER EXT	Steady—Spoilers deployed.	Steady—Normal when spoilers are extended.
	Flashing—Spoilers are extended with flaps > 3°°.	Flashing—Retract spoilers or flaps; normal on landing.

Table 4-1. ANNUNCIATORS (Cont)

WARNING/ CAUTION/ ADVISORY	PROBABLE CAUSE	PILOT ACTION
SPOILER MON	a. Spoiler/Aileron split > 6°. b. Spoiler/Spoiler split > 6°. c. AC or DC power failure to system. d. Internal computer fault.	Refer to SPOILER MON Light Illuminated in the “Abnormal Procedures” section of the *AFM*.
L STALL	Steady— a. Left stall system failed. b. Left stall circuit breaker is open. c. Indicator at 1.0	Steady— a. If one stall system fails, monitor other system for stall avoidance. If both systems fail, refer to Stall Warning System Failure in the “Abnormal Procedures” section of the *AFM*. b. Reset the left stall circuit breaker
	Flashing—Aircraft is in shaker range.	Flashing—Execute stall recovery procedure.
R STALL	Steady— a. Right stall system failed. b. Right stall circuit breaker is open. c. Indicator at 1.0	Steady— a. If one stall system fails, monitor other system for stall avoidance. If both systems fail, refer to Stall Warning System Failure in the “Abnormal Procedures” section of the *AFM*. b. Reset the right stall circuit breaker.
	Flashing—Aircraft is in shaker range.	Flashing—Execute stall recovery procedure.
PITCH TRIM	Ground/Airborne— a. Wheel master button is depressed. b. Trim speed monitor has detected a trim speed fault. c. Pitch trim selector switch is in OFF position. d. Attempting to trim using PRI TRIM SW with pitch trim selector in SEC position. e. Secondary pitch trim circuit breaker is out. Airborne— f. Mach monitor has detected a Mach trim computer output fault. Mach trim and pitch trim lights are both illuminated.	a. Release wheel master button. b,c,d, or e. Refer to PITCH TRIM L ight Illuminated in Flight” in the “Abnormal Procedures” section of the *AFM.* f. Refer to MACH TRIMand PITCH TRIM Lights Illuminated in the “Abnormal Procedures” section of the *AFM.*

Table 4-1. ANNUNCIATORS (Cont)

WARNING/ CAUTION/ ADVISORY	PROBABLE CAUSE	PILOT ACTION
L GEN	a. Generator switch is in OFF or START. b. Generator not on the line due to overvoltage or GPU plugged in. c. Generator failed. d. Generator circuit breaker is open.	Refer to Single Generator Failure in the “Abnormal Procedures” section of the *AFM* or Dual Generator Failure in the “Emergency Procedures” section of the *AFM*.
R GEN	a. Generator switch is in OFF or START. b. Generator not on the line due to overvoltage or GPU plugged in. c. Generator failed. d. Generator circuit breaker is open.	Refer to Single Generator Failure in the “Abnormal Procedures” section of the *AFM* or Dual Generator Failure in the “Emergency Procedures” section of the *AFM*.
BAT 160	One or both nicad batteries have reached 160°F.	Refer to Battery Overheat (Aircraft with Nickel-Cadmium Batteries) in the “Emergency Procedures” section of the *AFM*.
CABIN FIRE	Flashing—Smoke is detected in cabin baggage area. Steady—In test mode	Refer to Cabin/Cockpit Fire, Smoke, or Fumes and Cabin Fire Light Illuminated in the “Emergency Procedures” section of the *AFM*.
ENTRY DOOR	Steady—Both entry door handles are open. Flashing—One or more latch pins are not engaged, door handles are not fully locked, or the key lock is actuated.	Ground—Close door or have malfunction investigated. Do not take off with light illuminated. Airborne—Refer to ENTRY DOOR or AFT CABIN DOOR Light Illuminated in the “Emergency Procedures” section of the *AFM*.
EXT DOORS	One or more of the following external doors are not fully closed: • Tailcone baggage. • Tailcone access	Ground—Ensure all doors are properly secured before flight. Airborne—Refer to EXT DOORS Light Illuminated in the “Emergency Procedures” section of the *AFM*.
PRESS SYS	a. Positive differential pressure limit exceeded 9.8 PSID or negative 0.5 PSID. b. Cabin altitude exceeds limitation 8,600 feet in auto or 8,750 feet in manual. c. Pressurization controller detects a fault or electrical power is lost to the pressurization controller.	Refer to PRESS SYS LIGHT Illuminated in the “Abnormal Procedures” section of the *AFM*.

Table 4-1. ANNUNCIATORS (Cont)

WARNING/ CAUTION/ ADVISORY	PROBABLE CAUSE	PILOT ACTION
EMER PRESS	a. One or both emergency pressurization valves are in the emergency position. b. Either ENG FIRE T-handle is pulled while the corresponding bleed-air switch is on (60-001 through 60-015). c. Indicates an electrical fault which may prevent activation of emergency airflow.	a. Cabin altitude above 9,500 feet—Refer to Pressurization Loss at Altitude in the "Abnormal Procedures" section of the *AFM*. Cabin altitude below 9,500 feet—Refer to EMER PRESS Light Illuminated in the "Abnormal Procedures" section of the *AFM*. b. No action required if ENG FIRE T-handle is pulled.
PITOT HT	a. Either PITOT HEAT switch is OFF, or b. Current to either pitot heater is less than 3 amps with switches on.	Refer to Pitot Heat Light Illuminated Procedure in the "Abnormal Procedures" section of the *AFM*.
WING HT	Wing structure temperature has reached 215°F, wing heat ON or OFF, or wing leading edge is below 58°F decreasing or 70°F increasing when the wing heat is on.	Monitor wing temperature gage. Turn off stabilizer/wing heat for over-temp condition. Refer to Wing Heat Light Illuminated Procedure in "Abnormal Procedures" section of the *AFM*. The light may illuminate when the system is first turned on until the system warms up.
WSHLD OV HT	Windshield temperature has reached limit: Ground—250°F. Airborne—345°F. NOTE Auto feature should stop airflow and extinguish green WSHLD HT light until the amber WSHLD OV HT light extinguishes.	Normal—When airflow resumes, position WSHLD HT switch to OFF and then to HOLD when airflow is acceptable. Abnormal—If airflow continues with WSHLD OV HT light on, immediately move WSHLD HT switch to OFF. When WSHLD OV HT light extinguishes, use HOLD feature to control airflow. Refer to WSHLD OV HT Light Illuminated in the "Abnormal Procedures" section of the *AFM*.
BLEED AIR L	Overtemperature of engine bleed air in the pylon (250°F) or in bleed-air ducting that passes through the pylon (600°F) .	1. Move left bleed-air switch to OFF. 2. Refer to Bleed-Air Warning Light Illuminated in the "Emergency Procedures" section of the *AFM*.

Table 4-1. ANNUNCIATORS (Cont)

WARNING/ CAUTION/ ADVISORY	PROBABLE CAUSE	PILOT ACTION
BLEED AIR R	Overtemperature of engine bleed air in the pylon (250°F) or bleed-air ducting (600°F). NOTE If both left and right bleed-air lights illuminate simultaneously, the tailcone temperature sensor indicates an overheat (255°F).	1. Move right bleed-air switch to OFF. 2. Refer to Bleed-Air Warning Light Illuminated in the “Emergency Procedures” section of the *AFM*.
R ENG VIB	Indicates an abnormally high level of vibration in the associated engine.	1. Monitor Engine Instruments 2. Refer to Engine Vibration Light Illuminated in the “Abnormal Procedures” section of the *AFM*.
R ENG CHIP	Metal particles are detected in engine oil.	Refer to ENG CHIP Light Illuminated in the “Emergency Procedures” section of the *AFM*.
R FUEL PRESS	Less than 2.75 psi fuel pressure to engine-driven fuel pump. Light extinguishes at 3.75 psi.	Check that the jet pump switch is in the ON position and refer to FUEL PRESS Warning Light Illuminated in the “Emergency Procedures” section of the *AFM*.
R OIL PRESS	Right engine oil pressure is below 20 psi.	Refer to OIL PRESS Warning Light Illuminated in the “Abnormal Procedures” section of the *AFM*.
L ENG CMPTR	Either a hardware fault or a major software fault has occurred in the left engine FADEC system.	FADEC will select the healthiest channel to control the engines. On Ground—Do not dispatch the aircraft with an amber engine computer light illuminated. In Flight—If the amber ENG CMPTR and white ENG CMPTR lights for the left engine are illuminated, operate the thrust lever cautiously, and refer to Amber and White ENG CMPTR Light Illuminated in the “Abnormal Procedures” section of the *AFM*. Refer to Amber ENG CMPTR Light Illuminated in the “Abnormal Procedures” section of the *AFM*, if only the amber light illuminates.

Table 4-1. ANNUNCIATORS (Cont)

WARNING/ CAUTION/ ADVISORY	PROBABLE CAUSE	PILOT ACTION
L ENG CMPTR	A minor software fault has occurred in the system.	FADEC will select the healthiest channel to control the engines. Refer to White ENG CMPTR Light in the "Abnormal Procedures" section of the *AFM*. Dispatch is not permitted.
L ENG FILTERS	Indicates impending bypass of engine fuel filter, airframe-mounted fuel filter or oil filter on left engine.	Refer to ENG FILTERS Light Illuminated in the "Abnormal Procedures" section of the *AFM*. Investigate problem on ground prior to next flight.
TR DEPLOY	Indicates thrust reverser is fully deployed.	a. Normal—Stow thrust reverser after deployment for normal operation. b. Abnormal—If TR inadvertently deploys, refer to Inadvertent Thrust Reverser Deployment During Takeoff in the "Emergency Procedures" section of the *AFM*.
APR ON	Automatic Performance Reserve has automatically activated while APR switch in ARM position.	Select OFF position on APR control switch when APR is no longer required.
LOW FUEL	Fuel level in either wing tank is less than 410 pounds.	Transfer fuel, refer to LOW FUEL Light Illuminated in the "Abnormal Procedures" section of the *AFM*. NOTE Wing electric scavenge pump(s) will be automatically energized when the low-fuel light illuminates.
SPOILER ARM	The spoiler lever has been placed in the ARM position.	Place spoiler lever to RET after takeoff or to EXT at landing touchdown.
STEER ON	Nosewheel steering is engaged by arm switch or either wheel master button and ground speed is less than 90 knots.	If engaged with the arm switch, no action is required. If engaged by the wheel master, release the button when nosewheel steering is no longer needed.
CABIN AIR	Indicates that any of the following switches are in the OFF position: a. L BLEED AIR b. R BLEED AIR c. CAB AIR	Select L & R BLEED AIR and CAB AIR ON prior to takeoff.

Table 4-1. ANNUNCIATORS (Cont)

WARNING/ CAUTION/ ADVISORY	PROBABLE CAUSE	PILOT ACTION
MACH TRIM	Electrical failure to Mach trim computer, the Mach trim monitor has deactivated the system, or Mach monitor has deactivated the system (PITCH TRIM light also illuminated in the latter case).	Reduce below 0.77 MI; or engage autopilot and refer to Mach Trim Malfunction in the "Abnormal Procedures" section of the *AFM*.
T.O. TRIM	Ground—Pitch trim is not within range for a safe takeoff.	Move pitch trim to takeoff segment.
CUR LIM	One or both 275 amp current limiters are blown.	On Ground—Refer aircraft to maintenance to repair fault and replace current limiters. Airborne—Refer to CUR LIM Light Illuminated During Flight in the "Abnormal Procedures" section of the *AFM*.
ELEC PWR	Left or right DC amps or volts have reached the caution or warning range; or, left or right AC volts have reached the caution or warning range.	Find which failure has occurred by looking at power monitor, and refer to ELEC PWR Light Illuminated in the "Abnormal Procedures" section of the *AFM*.
BAT 140	One or both nicad batteries have reached 140°F.	Refer to Battery Overheat (Aircraft with Nickel-Cadmium Batteries) in the "Emergency Procedures" section of the *AFM*.
INSTR FAN	The instrument fan is off.	Investigate the problem on the ground prior to next flight.
AFT CAB DOOR	Steady—Aft cab door handle open. Flashing—One or more latch pins are not fully engaged, door handle is not fully closed, or the ground security pin is installed.	Ground—Close door or have malfunction investigated. Do not takeoff with light illuminated. Airborne—Refer to Entry Door or Aft Cabin Door Light Illuminated in the "Emergency Procedures" section of the *AFM*.
CABIN ALT HI	Cabin altitude has exceeded 10,000 ft (high altitude warning horn sounds).	Perform emergency descent. Horn may be silenced for 60 seconds with mute button in right throttle.

Table 4-1. ANNUNCIATORS (Cont)

WARNING/ CAUTION/ ADVISORY	PROBABLE CAUSE	PILOT ACTION
DUCT OV HT	Temperature in the cabin or crew bleed-air distribution ducting in tailcone has reached approximately 300°F.	Check temperature control valve for affected system (cab or crew) in full cold position. If it isn't, select manual and full cold for affected system. Refer to Duct OV HT Light Illuminated in the "Abnormal Procedures" section of the *AFM*.
ICE DET	If the optional ice detection probe is installed: • Icing conditions exist and the STAB HEAT switch is turned ON.	No action required—advisory only.
ICE DET	If the optional ice detection probe is installed… • Icing conditions exist and the stab heat switch is turned off, or • There is an internal fault in the ice detection box.	If icing conditions exist, turn the anti-ice systems on.
STAB HT	Ground (Steady)—Normal with stab/wing heat switch on. STAB HEAT is disabled by squat switch (should be in test mode).	Place STAB WING HEAT switch to OFF if not required.
	Ground/Airborne (Flashing)—STAB HEAT failed self-test on ground or one or more elements failed in flight.	Refer to STAB HT Light Illuminated in Flight in the "Abnormal Procedures" section of the *AFM*.
	Airborne (Steady) Abnormal—System has failed or power to STAB HEAT has failed.	
WSHLD HT	Windshield anti-ice system is on.	Select OFF position of WSHLD HT switch when not required for anti-ice or rain removal.
L WS DEFOG	a. The system has been activated and the temperature is still below 80°F. b. The windshield temperature is above 150°F. c. Sensor is open and/or windshield has reached temperature limit and/or loss of required power.	a. Normal operation. Feel the left windshield near the crew member's head. b. If an underheat condition exists, no action is required. c. If an overheat condition exists, pull the L WSHLD Defog circuit breaker. Refer to WS DEFOG Light Illuminated in the "Abnormal Procedures" section of the *AFM*.

Table 4-1. ANNUNCIATORS (Cont)

WARNING/ CAUTION/ ADVISORY	PROBABLE CAUSE	PILOT ACTION
R WS DEFOG	a. The system has been activated and the temperature is still below 80°F. b. The windshield temperature is above 150°F. c. Sensor is open and/or windshield has reached temperature limit and/or loss of required power.	a. Normal operation. Feel the right windshield near the crew member's head. b. If an underheat condition exists, no action is required. c. If an overheat condition exists, pull the L WSHLD Defog Circuit Breaker. Refer to WS DEFOG Light Illuminated in the "Abnormal Procedures" section of the *AFM*.
TR DEPLOY	Indicates thrust reverser is fully deployed.	a. Normal—Stow thrust reverser after deployment for normal operation. b. Abnormal—If TR inadvertently deploys, refer to Inadvertent Thrust Reverser Deployment During Takeoff in the "Emergency Procedures" section of the *AFM*.
R ENG FILTERS	Indicates impending bypass of engine fuel filter, airframe-mounted fuel filter or oil filter on right engine.	Refer to ENG FILTERS Light Illuminated in the "Abnormal Procedures" section of the *AFM*. Investigate problem on ground prior to next flight.
R ENG CMPTR	A minor software fault has occurred in the system.	FADEC will select the healthiest channel to control the engines. Refer to White ENG CMPTR Light in the "Abnormal Procedures" section of the *AFM*. Dispatch is not permitted.
R ENG CMPTR	Either a hardware fault or a major software fault has occurred in the right engine FADEC system (one channel only).	FADEC will select the healthiest channel to control the engines. On Ground—Do not dispatch the aircraft with an amber engine computer light illuminated. In Flight—If the amber ENG CMPTR and white ENG CMPTR lights for the right engine are illuminated operate the thrust lever cautiously, and refer to Amber and White ENG CMPTR Light Illuminated in the "Abnormal Procedures" section of the *AFM*. Refer to Amber ENG CMPTR Light Illuminated in the "Abnormal Procedures" section of the *AFM*, if only amber light illuminates.

Table 4-1. ANNUNCIATORS (Cont)

WARNING/ CAUTION/ ADVISORY	PROBABLE CAUSE	PILOT ACTION
TR UNLOCK	Thrust reverser is not fully stowed or deployed.	Insure thrust reverser becomes fully deployed or stowed. In Flight—Refer to TR UNLOCK Light Illuminated in the “Abnormal Procedures” section of the *AFM*.
APR ARM	APR system is armed.	Select OFF position on APR control switch when APR is no longer required.
FUEL SYS	Steady— a. Any transfer/fill operation is in progress. b. Crossflow valve selected open. c. Standby pump(s) selected on. Flashing—Fuselage tank full or empty.	Steady—a, b, or c. Monitor the appropriate system as required. Flashing—Discontinue fuel transfer or fill operation.
L NAC HT	a. If nacelle heat switch is on, bleed-air pressure to the nacelle lip is insufficient for anti-icing or the valve for stator vane heating is not full open. b. If nacelle heat switch is off, bleed-air pressure is present in the nacelle lip due to a malfunctioning valve or the valve for stator vane heating is not full closed.	a. Refer to NAC HT Light Illuminated in the “Abnormal Procedures” section in the *AFM*. b. If the left engine is at idle, advance power lever to extinguish the light. c. If nacelle heat is not obtained, avoid flight into icing conditions.

Table 4-1. ANNUNCIATORS (Cont)

WARNING/ CAUTION/ ADVISORY	PROBABLE CAUSE	PILOT ACTION
R NAC HT	a. If nacelle heat switch is on, bleed-air pressure to the nacelle lip is insufficient for anti-icing or the valve for the stator vane heating is not fully open. b. If nacelle heat switch is off, bleed-air pressure is present in the nacelle lip due to a malfunctioning valve or the valve for the stator vane heating is not fully closed.	a. Refer to NAC HT Light Illuminated in the "Abnormal Procedures" section of the *AFM*. b. If the right engine is at idle, advance power lever to extinguish the light. c. If nacelle heat is not obtained, avoid flight into icing conditions.
TR UNLOCK	Thrust reverser is not fully stowed or deployed.	Insure thrust reverser becomes fully deployed or stowed. In Flight—Refer to TR Unlocked Light Illuminated in the "Abnormal Procedures" section of the *AFM*.
TR ARM	Normal—Thrust reverser circuit breakers are set, aircraft is on the ground (squat switch in ground mode), and corresponding thrust lever at idle.	None required.
	Abnormal—Flashing TR Arm Light in flight.	Refer to the TR ARM Light in Flight in the "Abnormal Procedures" section in the *AFM*.
EDS FAULT	a. EDS has lost power. b. EDU bite has detected a system failure. c. EDU memory is 85% full. d. System has detected an engine condition which is out of acceptable parameters.	Refer to EDS FAULT Light Illuminated in the "Abnormal Procedures" section in the *AFM*. Notify maintenance of the fault.
ENG SYNC	Engine sync is ON and nose landing gear is not up and locked.	Turn engine synchronization off when gear is down for takeoff or landing. Also, turn engine synchronization off during single-engine operation.
NAC HT ON	The left or right nacelle heat switch is on.	a. Turn nacelle heat off if not in icing conditions. b. Comply with Ground Operation Limitations in the "Limitations Section" of the *AFM*.

Table 4-1. ANNUNCIATORS (Cont)

WARNING/ CAUTION/ ADVISORY	PROBABLE CAUSE	PILOT ACTION
ALC LOW	Windshield alcohol tank empty.	Refer to ALC Low Light Illuminated in the "Abnormal Procedures" section of the *AFM*. Select OFF position on WSHLD ALC switch when alcohol stops flowing. Have tank serviced before next flight.
TR ARM	Normal—Thrust reverser circuit breakers are set, aircraft is on the ground (squat switch in ground mode), and corresponding thrust levers at idle.	None required.
	Abnormal—Flashing TR arm light in flight.	Refer to TR ARM Light in Flight in the "Abnormal Procedures" section of the *AFM*.
ENG FIRE PULL	A fire or an overheat condition exists in the affected engine.	Refer to Engine Fire – Shutdown in the "Emergency Procedures" section of the *AFM*.
ENG EXT ARMED	An engine fire T-handle has been pulled and fire extinguishing bottle is ready for use/discharge.	If a fire or overheat condition exists, depress the light to allow the contents of one fire extinguishing bottle to flow into the associated nacelle.
RB	Failure of the rudder boost self-test sequence. Rudder boost is unable to deliver servo torque. Avionics master switches are off.	Check to ensure the rudder boost switch is on and avionics master switches are on.
RB	Pilot is depressing either rudder with at least 50 lbs. of pressure and rudder boost is delivering servo torque.	No action required—indicates normal RB operation when heavy rudder pressure is required.
TRIM	The autopilot is unable to activate pitch trim.	Refer to Autopilot FCP TRIM Light Illuminates in Flight in the "Abnormal Procedures" section of the *AFM*.

Table 4-1. ANNUNCIATORS (Cont)

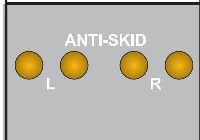
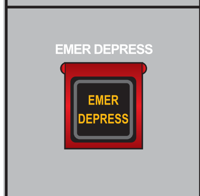
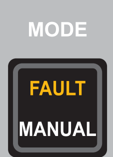
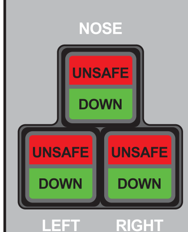
WARNING/ CAUTION/ ADVISORY	PROBABLE CAUSE	PILOT ACTION
	A fault has been detected in the antiskid circuitry. If all four lights are illuminated, power to the control box may have been lost, or the antiskid switch is off.	Cycling the antiskid switch off, then on, may clear the fault. Apply brakes judiciously and modulate brake pressure to avoid skidding the tires. Refer to the “Abnormal Procedures” section of the *AFM.*
	Emergency depressurization has been manually selected, and the outflow valves have been opened.	Cabin pressure will reduce to equal the pressure outside the aircraft up to 13,700 feet.
	Power is being used from the emergency batteries, and they are not receiving a trickle charge from the aircraft’s electrical system.	Refer to Dual Generator Failure in the “Emergency Procedures” section of the *AFM.*
	Amber FAULT light indicates there has been a power loss, cabin altitude has exceeded 8,600 feet, or a fault has been detected in the pressurization module. PRESS SYS annunciator will also be illuminated.	When FAULT light illuminates, the system automatically reverts to manual mode. See PRESS SYS Light Illuminated in the “Abnormal Procedures” section of the *AFM.*
	The MANUAL light illuminates when manual has been selected with mode switch.	If MAN is selected while a FAULT light and PRESS SYS annunciator are illuminated, they will extinguish. If MAN is selected when no fault is indicated, manual mode is activated. Depressing the MAN switch a second time will cause “MAN” to extinguish and return the system to the automatic mode.
	The green DOWN lights indicate the associated landing gear is locked down.	If one or more lights are off, and an attempt has been made to cycle the gear down, refer to Alternate Gear Extension/Electrical Malfunction in the “Abnormal Procedures” section of the *AFM.*
	The nose gear red unsafe light is on when the nose gear is in transit. The two main gear red unsafe lights are on when the inboard gear doors are not locked up.	If one or more lights remain on after cycling the gear, refer to Alternate Gear Extension/Electrical Malfunction in the “Abnormal Procedures” section of the *AFM.* NOTE The inboard gear doors will not lock up after using either alternate gear extension system (Blowdown or Free Fall).

Table 4-1. ANNUNCIATORS (Cont)

WARNING/ CAUTION/ ADVISORY	PROBABLE CAUSE	PILOT ACTION
IGNITION	The respective engine air ignition ignitor box is receiving power.	Normal operation.
LO FUEL PRESS	Low pressure exists in the engine fuel supply line. LO FUEL PRESS annunciation is also on the Fuel Control Panel.	Check that the jet pump switch is in the ON position and refer to FUEL PRESS Warning Light Illuminated in the Emergency Procedures Section of the *AFM*.
MUTE	The gear warning horn has been silenced by depressing either gear warning horn mute buttons.	Normal operation.
PARK BRAKE	The parking brake handle is pulled.	Position the brake handle as desired.
START START	The respective starter is engaged.	If the starter does not disengage after start, move the START-GEN switch to GEN. If the light remains illuminated while on the ground, shut down the affected engine. See Starter Engaged Light Remains Illuminated After Start in the "Abnormal Procedures" section in the *AFM*.
WARN CAUT	The red WARNING portion of the light indicates a red light is illuminated on the glareshield or panel. The amber CAUTION portion of the light illuminates when amber light has illuminated on the glareshield or panel.	Pilot action depends on which glareshield/panel lights have illuminated. Depressing a WARN/CAUT light will cause both WARN/CAUT lights to extinguish.
ARM NOSE STEER	Nose gear is down and locked, nose steering is armed.	Use nose steering on ground below 90 knots.
CAT II (amber)	Category II enabled and monitored parameters NOT met (amber).	Prior to FAF–determine and correct. After FAF–revert to Category I minimums.
CAT II (green)	Category II enabled and monitored parameters met.	None.
DISPLAY TEMP	The temperature warning sensor has detected an approaching thermal shutdown and the ground/sky raster is removed to reduce power requirements of the display.	Select PFD REV (reversion) on the EFIS control panel.

QUESTIONS

1. How can all glareshield annunciator lights be tested?

A. By individual system testing
B. By depressing each individual capsule
C. By depressing either TEST switch
D. By shutting the represented system OFF

2. When a red glareshield light illuminates, what other annunciation occurs?

A. Only the pilot master warning light flashes
B. Both master warning lights illuminate steady
C. Only the copilot master warning light illuminates steady
D. Both master warning lights flash

3. If an illuminated amber or red glareshield light suddenly extinguishes, what does it indicate?

A. Five minutes have passed.
B. The malfunction no longer exists.
C. Three minutes have passed.
D. The master warning lights have been reset.

4. When an amber glareshield light illuminates, what other annunciation occurs?

A. Only the pilot master caution light flashes
B. Both master caution lights illuminate steady
C. Only the copilot master caution light illuminates steady
D. Both master caution lights flash

5. How is the intensity adjusted on the glareshield annunciator?

A. Automatically by photo electric cells
B. By depressing the TEST button
C. By depressing each individual capsule
D. By depressing the DIM button

CHAPTER 5
FUEL SYSTEM

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ILLUSTRATIONS

TABLE

CHAPTER 5
FUEL SYSTEM

INTRODUCTION

The Learjet 60 fuel system consists of the fuel storage, quantity indicating, transfer, distribution, vent, and single-point pressure refueling systems.

The fuel system is covered in this chapter from the storage areas to the high-pressure engine fuel pumps, at which point fuel system operation becomes a function of the powerplant. Refer to Chapter 7—“Powerplant.”

GENERAL

The fuel system provides for fuel storage and low-pressure fuel distribution to the high-pressure engine fuel pumps.

The fuel storage system consists of an integral wet-wing tank in each wing, a fuselage tank, and a vent system.

The fuel quantity indicating system utilizes capacitance-type probes to measure fuel quantity in each tank.

A crossflow valve permits transfer of fuel between wings for fuel balancing. It is also open during transfer of fuel, either from wings to fuselage or from fuselage to wings. Each wing contains an electrical scavenge pump and three transfer jet pumps to automatically move fuel to the lowest point of the wing tank. A main, engine supply jet pump and an electric standby pump are located at the low point of the wing tank to pick up and deliver low-pressure fuel to the respective engine fuel

pump. The engines are fed directly from the respective wing tanks. Fuselage tank fuel must be transferred forward to the wings before it can be fed to the engines.

In-flight fuel transfer is from the fuselage tank to the wing tanks (Figure 5-1).

The left (NORM XFR) fuel transfer system moves fuel forward through the left transfer valve and line to the left wing, and to the right wing through the crossflow valve.

The right (AUX XFR) fuel transfer system moves fuel forward through the right transfer valve and line to the right wing, and to the left wing through the crossflow valve. The two transfer systems may be used separately or simultaneously. When both are used simultaneously, it is called rapid transfer. The gravity transfer system may also be used to gravity transfer fuel from the fuselage to both wing tanks.

When flying in an extreme noseup attitude for extended periods, fuel may gravity-flow from the wing tanks to the fuselage tank. Periodically transfer fuel back into the wings.

The ram-air vent system affords slight pressurization to the fuel tanks in flight. On the ground, the system is a straight atmospheric vent system.

Drain valves enable draining of condensation and contaminants which have settled in the vent sumps.

A fuel filler is provided for each wing tank (Figure 5-1). The wings may be filled through the over wing ports. The fuel may then be transferred to the fuselage tank through fill operations. A single-point pressure refueling system is installed, which allows pressure refueling to the wing and fuselage tanks.

The flight crew must ensure that only approved fuels and anti-icing additives are used as specified in the Learjet 60 *AFM*.

WING TANKS

Each wing tank extends from the aircraft centerline rib to a point just short of the winglets, with the exception of the following areas: the main landing gear wheel well, the leading edge forward of spar 1 (wing heat area), and the trailing edge between spar 7 and 8 (flap, spoiler, and aileron areas).

The 2.5° wing dihedral makes the inboard portions of the wing tanks the lowest areas. The main jet pumps and electric standby pumps, which supply fuel to the engines, are located in the low areas and remain submerged in fuel until the tanks are nearly empty (Figure 5-1).

Wing tanks are baffled with ribs and spars to minimize fuel shift. Flapper-type check valves, located in the wing ribs, allow unrestricted inboard flow of fuel and inhibit outboard flow. Two pressure-relief valves at the centerline rib allow fuel or air to flow between tanks if pressure in one wing tank is greater than the other.

The useable fuel capacity of each wing tank is approximately 1,449 pounds (216.4 gallons) if fuel tanks are filled through the wing fillers or 1,400 pounds (209 gallons) if filled through the single-point pressure refueling system.

FUSELAGE TANK

The fuselage tank is two bladder-type fuel cells located aft of the rear pressure bulkhead. The useable fuel capacity of the fuselage tank is approximately 5,012 pounds (748 gallons) when completely filled. The tank can be filled by either the single point pressure refueling system (5,000 pounds) or by transferring fuel from the wings to the fuselage tank (5,012 pounds).

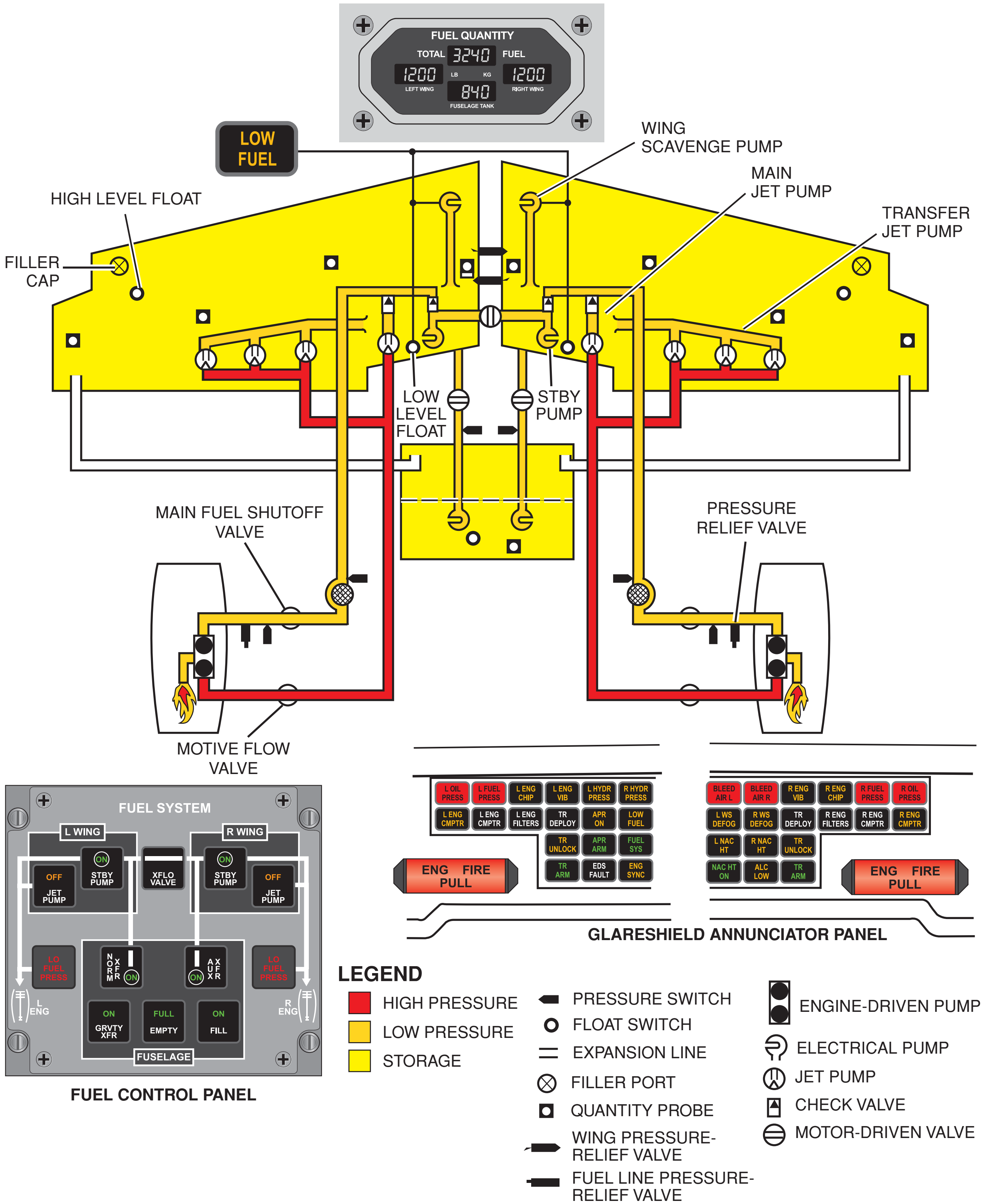


Figure 5-1. Fuel System

WING AND SINGLE-POINT PRESSURE REFUELING FILLER PORTS

A fuel filler port is located on the outboard portion of each wing (Figure 5-2). A single-point pressure refuel system is installed with a fill valve in each tank. The SPPR filler connection and control panel are on the right side of the fuselage just aft of the wing trailing edge and below the engine pylon (Figure 5-2).

FUEL QUANTITY INDICATING SYSTEM AND CONTROLS

The capacitance-type quantity indicating system consists of two processors in the panel-mounted indicator, nine capacitance probes, and a fuel density compensation probe (Figure 5-3).

One probe is located in the fuselage tank and four probes are located in each wing tank (see Figure 5-1). The inboard probe in the left wing contains a temperature compensator which compensates for fuel density changes due to

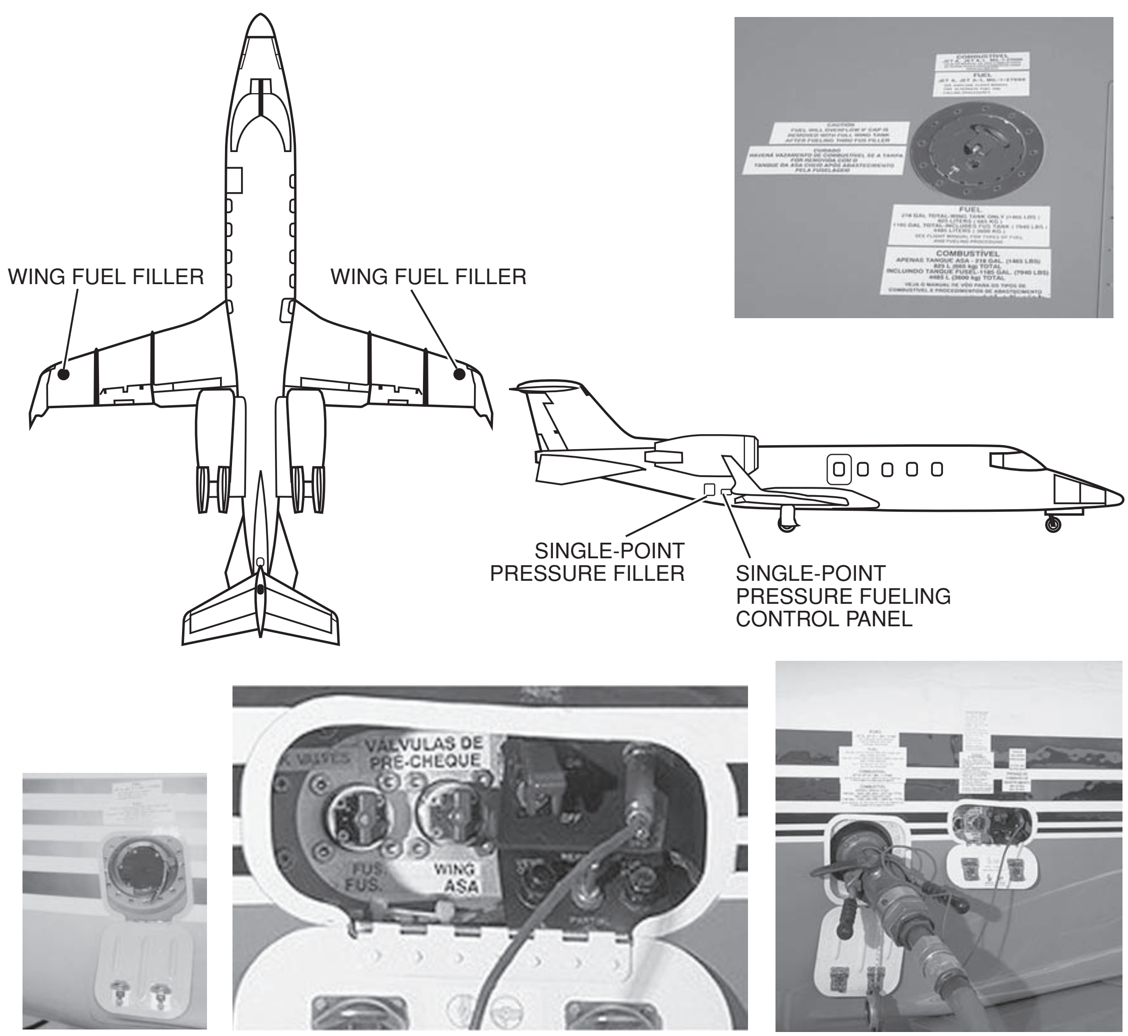
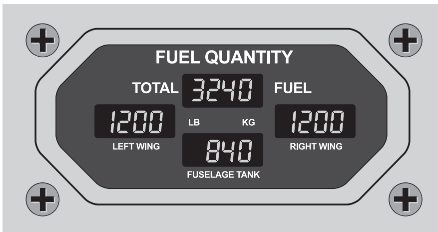


Figure 5-2. Wing and Fuselage Tank Fuel Filler Ports and Locations



COCKPIT INDICATOR

Figure 5-3. Fuel Quantity Indicators

temperature. The indicator applies the density correction to all the quantity indications (wings, fuselage, and total).

There are two processors in the indicator. One supplies data to the left wing and total quantity, the second processor supplies data to the right wing and fuselage tank. A failure of one processor will not cause complete loss of quantity indication.

The fuel quantity indicating system uses DC supplied through the FUEL QTY PWR 1 and 2 circuit breakers located in the pilot and copilot fuel group of circuit breakers. Either circuit breaker will supply power to the processors and indicator, giving the system a redundant power source in the event of electrical failure. Both circuit breakers are on the emergency buses. Both circuit breakers must be pulled and reset in order to reset either processor. The indicator provides a continuous digital readout of fuel remaining in each wing tank, fuselage tank, and total remaining useable fuel.

The indicator will also flash the digital display of a heavy wing and an IMB display if a fuel imbalance is detected. The imbalance must be present for 30 seconds before annunciating.

The indicator will flash the heavy wing indication with a 200-pound IMB display imbalance when the flaps are set to 8° or lower. A 500-pound imbalance will flash the heavy wing indication and IMB display with the flaps up. The flashing indication may be cancelled by depressing the mute switch in the right thrust lever. The IMB display will remain illuminated until the imbalance is corrected below the trip point.

The fuel indicating system has built-in test capabilities to monitor the state of the fuel quantity indicating system including the indicator, probes and wiring. When failures are detected for longer than 45 seconds, an error code will be displayed corresponding to the particular error found. System and indicator faults will be displayed in the TOTAL display, whereas specific tank faults will be displayed on the corresponding tank display. For example, Error Code 5 (ER-5) indicates possible fuel contamination. See Table 5-1 for list of error codes.

Table 5-1. FUEL SYSTEM ERROR CODES

ERROR CODE #	DESCRIPTION
0	Internal indicator failure such as a interprocessor communications, memory test failures, or gauging hardware failures
1	Probe capacitance out of range
2	Tank probe/line grounded or open
3	Compensator capacitance out of range
4	Grounded or open compensator
	Probe/line
5	Fuel contamination
6	Loss of one DC power input, available on system initialization only
7	Tank calibration parameters incorrect
8	Loss of aircraft pitch information *(on aircraft 60-159 and subsequent and prior aircraft modified by SB 60-28-10)*
9	Oscillator failure
A	Reference failure

The cockpit indicator is tested using the glareshield warning light test. The digits read 8880 (in all four windows) for a proper test.

Indicator illumination intensity can be controlled through the CENTER PNL PEDESTAL lighting control on the pilot dimmer panel (lower left instrument panel).

LOW FUEL LIGHT

The low-level float switches (one in each wing) trip when fuel quantity in the associated wing tank decreases to approximately 410 pounds (61.2 gallons) and illuminate the single amber LOW FUEL light on the pilot glareshield annunciator panel. The low level must be present for 30 seconds before the LOW FUEL light annunciates. The light may be affected by pitch

attitude. In nosedown attitudes the light may illuminate at a greater quantity. Either float switch may cause the light to illuminate. At the same time, when a low-level float switch trips, the on side electrical scavenge pump is automatically energized. They pickup fuel from the bottom of the forward wing tank sections and move it to the area in the tanks where the main jet pumps are located. If the left wing low float switch trips the LOW FUEL light while the fuselage fill function is ON, the fuselage fill function will trip OFF.

The scavenge pumps are powered through the L and R STBY-SCAV pump in the FUEL group of circuit-breaker panels on the left and right circuit-breaker panels. They are not powered during EMER bus operation.

Operation of the scavenge pumps may be checked prior to engine start by depressing either ANNUNCIATOR TEST switch on the glareshield annunciator panel and listening for audible indication.

FUEL TRANSFER SYSTEM

The fuel transfer system consists of those components of the fuel system related to moving fuel between tanks.

The components in the fuel system utilized to supply fuel to the engines are covered in this chapter under engine fuel supply system.

CROSSFLOW VALVE

The crossflow valve is installed in the crossflow line connecting the two wing tanks (see Figure 5-1). It is normally closed during flight, but it is opened during fuselage fuel transfer and may be opened when necessary to move fuel from one wing tank to the other.

This motor-driven valve is controlled by the XFLO VALVE switch on the fuel control panel (see Figure 5-1) and is powered through the XFLO VALVE circuit breaker on the pilot fuel circuit-breaker panel. The crossflow valve is operative during emergency bus mode. It also opens anytime GRVTY XFR, NORM XFER, AUX XFR, or FILL is selected. A horizontal bar on the switch will illuminate to annunciate the valve's open status and will not be illuminated when the valve is closed. If the crossflow valve is neither open or closed, the horizontal bar will flash.

The green FUEL SYS light on the glareshield annunciator panel (see Figure 5-1) will illuminate steady whenever the crossflow valve is open, a standby pump is on, or any transfer/fill operation is in progress. The light will flash when the fuselage tank is full and FILL is selected or when the tank is empty and normal and/or auxiliary fuel transfer is selected.

FUEL VALVES

The indicator lights for the fuel valves (crossflow, transfer, and jet pump/motive flow) are valve position indicator lights (open, closed or in between). If the valve fails between open or closed, its indicator light will flash. The indicator lights will also flash for a short period of time while the valve is in transit to a selected position.

STANDBY PUMPS

The standby pumps are located next to the main jet pumps (see Figure 5-1) at the low point in each wing tank. The standby pumps are used:

- For engine start (automatically energize with starter switch activation)
- As a backup for main jet pumps
- For wing-to-wing crossflow
- For wing-to-fuselage transfer of fuel (automatically energized with the FILL switch on)

The standby pump in each wing is powered through the L or R STBY-SCAV PUMP circuit breaker located in the pilot/copilot FUEL group of circuit breakers. These pumps are controlled by switches on the fuel panel (STBY PUMP, FILL) (see Figure 5-1). The standby pumps are both disabled whenever any of the fuselage

transfer (fuselage tank to wing) systems are activated. Also, the standby pumps do not operate in the EMER bus mode of operation.

TRANSFER LINES AND VALVES

Two transfer lines (one on each side) connect the fuselage tank and the wing tanks. A transfer valve is installed in each line (see Figure 5-1). The transfer valves control fuel movement between the fuselage and wing tanks.

Fuel can be transferred from the fuselage tank to both wings through either the LH or RH transfer lines or through both lines at the same time. Fuel can also be pumped from the wings to the fuselage tank through the LH and RH transfer lines. These operations are controlled by switches on the fuel control panel labeled: GRVTY XFR, NORM XFR, AUX XFR, and FILL (see Figure 5-1). Use of these switches is covered in Section II, Fuel Management, in the Learjet 60 *Airplane Flight Manual*.

The transfer valves are electrical motor-driven valves powered through the respective L and R JET PUMP–XFR VALVE circuit breakers located on the pilot/copilot FUEL group of circuit breakers. The valves will remain in the last selected position if power to them is lost. Valve position is indicated on the fuel panel and by the green FUEL SYS light on the glareshield annunciator. With a valve open, the vertical bar is illuminated, and if the valve is closed, the bar is not illuminated. If the valve is in between open and closed, the bar will flash.

TRANSFER (FUSELAGE) PUMPS

The left and right transfer pumps are electrical motor-driven pumps located at the low point in the fuselage tank. The left transfer pump is used for the NORM XFR system. The right transfer pump is used for the AUX XFR system.

The flow rate is approximately 50 pounds a minute through each transfer line.

The left and right transfer pumps are powered through the FUS TANK XFR PUMP and the FUS TANK AUX PUMP circuit breakers located in the pilot/copilot FUEL group of circuit breakers. Both transfer pumps remain operative during emergency bus mode operation.

PRESSURE SWITCHES

A pressure switch is installed between the transfer pump and transfer valve in each transfer line to indicate completion of fuel transfer from the fuselage tank to the wings. Either press switch (or both pressure switches) will illuminate the white EMPTY light on the fuel panel when fuel pressure is lost in the respective transfer line. The FUEL SYS annunciator will also flash.

Pressure switch circuitry for the left pressure switch is only energized when NORM XFR is selected. The right pressure switch is only energized when AUX XFR is selected.

FLOAT SWITCHES

Wing High-Level Float Switches

During normal transfer (LH transfer system), fuel transfer will continue until a high-level float switch (one in the outboard end of each wing tank) senses a wing full condition. Either float switch will automatically deenergize the left fuselage pump and close the left transfer valve when either wing becomes full. If the NORM XFR switch is left ON, transfer resumes when both high-level float switches sense that the wing tanks are not full. This automatic cycling continues until the fuselage tank is empty. Gravity transfer and auxiliary transfer are not affected by the wing high-level float switches.

Fuselage Tank Float Switch

This float switch provides indication that the fuselage tank is full during filling of the fuselage tank from the wings (FILL switch is ON) and automatically terminates the fill operation.

Wing Tank Low-Level Float Switch

See discussion of low fuel light, this chapter.

FUSELAGE FUEL TRANSFER—FILL OPERATION AND WING-TO-WING CROSSFLOW

See Section II, Fuel Management, of the *Airplane Flight Manual* for operation procedures. When discussing fuselage fuel transfer-fill and wing-to-wing crossflow, use Figures 5-4 through 5-9 as a guideline.

NOTES

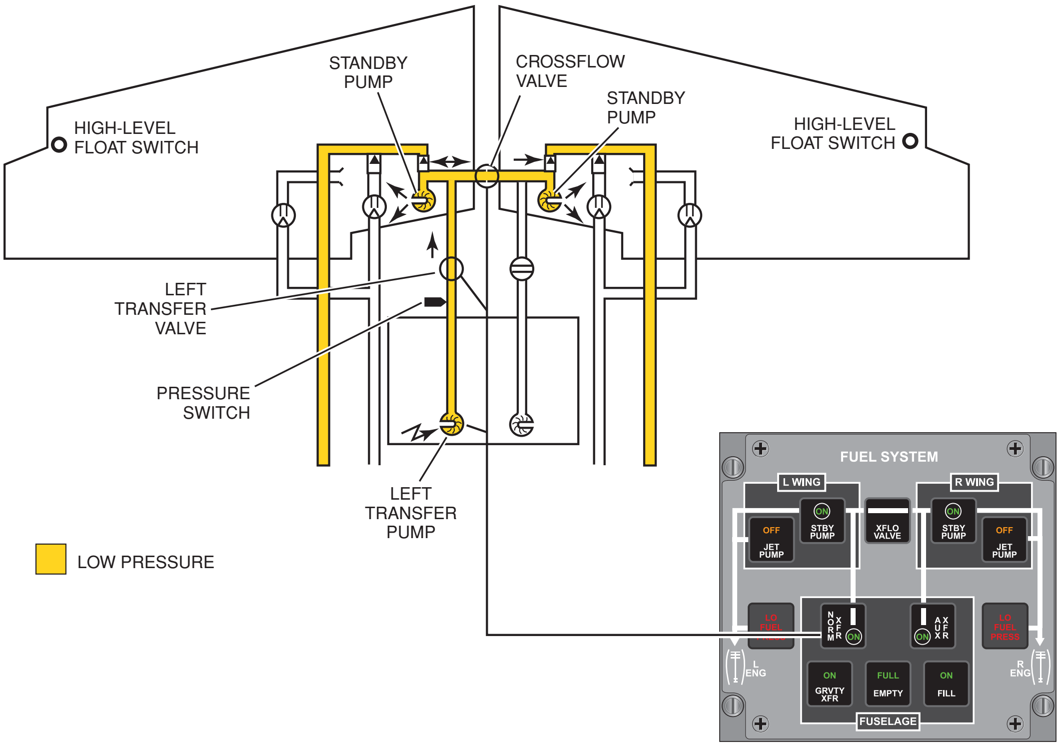


Figure 5-4. Fuselage Tank-to-Wing Tanks (Normal Transfer)

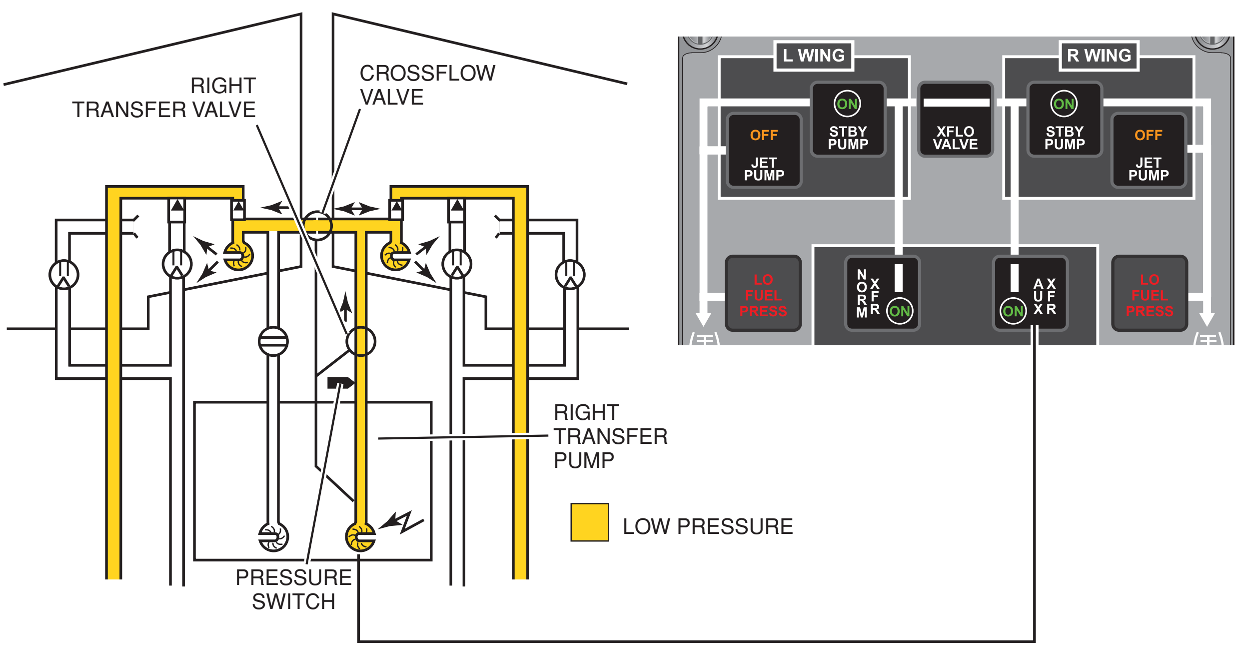


Figure 5-5. Fuselage Tank-to-Wing Tanks (Auxiliary Transfer)

5 FUEL SYSTEM

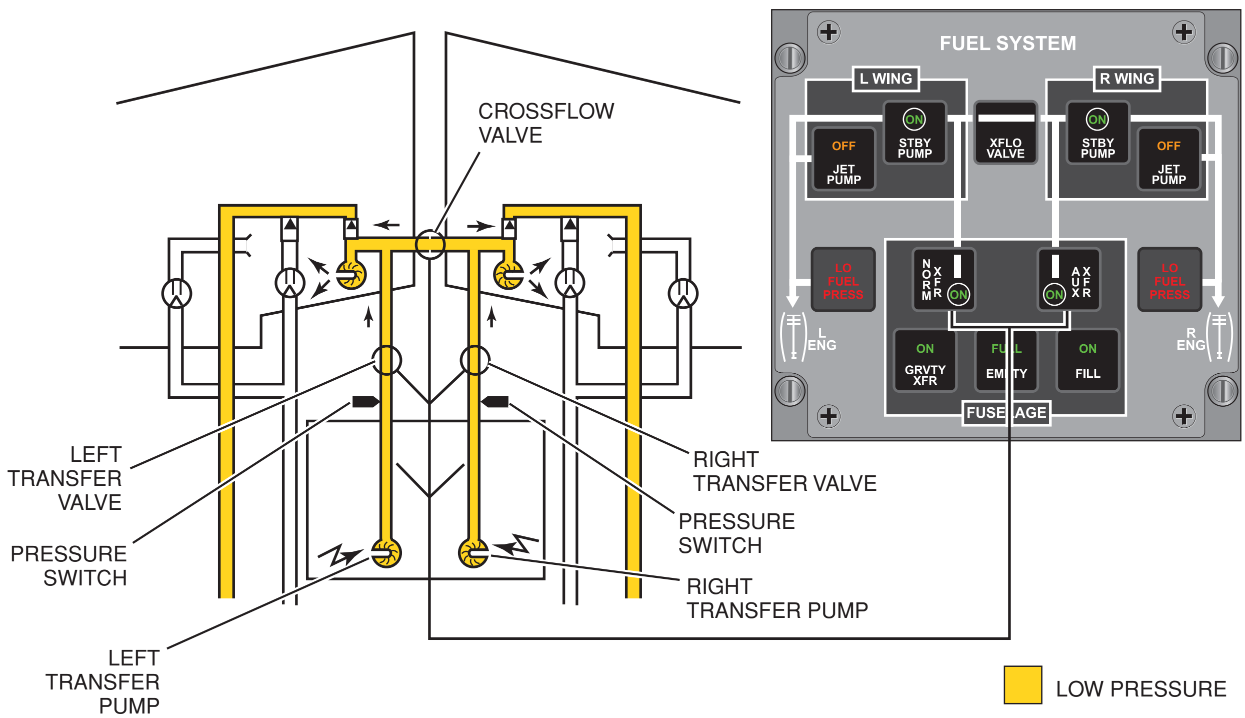


Figure 5-6. Fuselage Tank-to-Wing Tanks (Rapid Transfer)

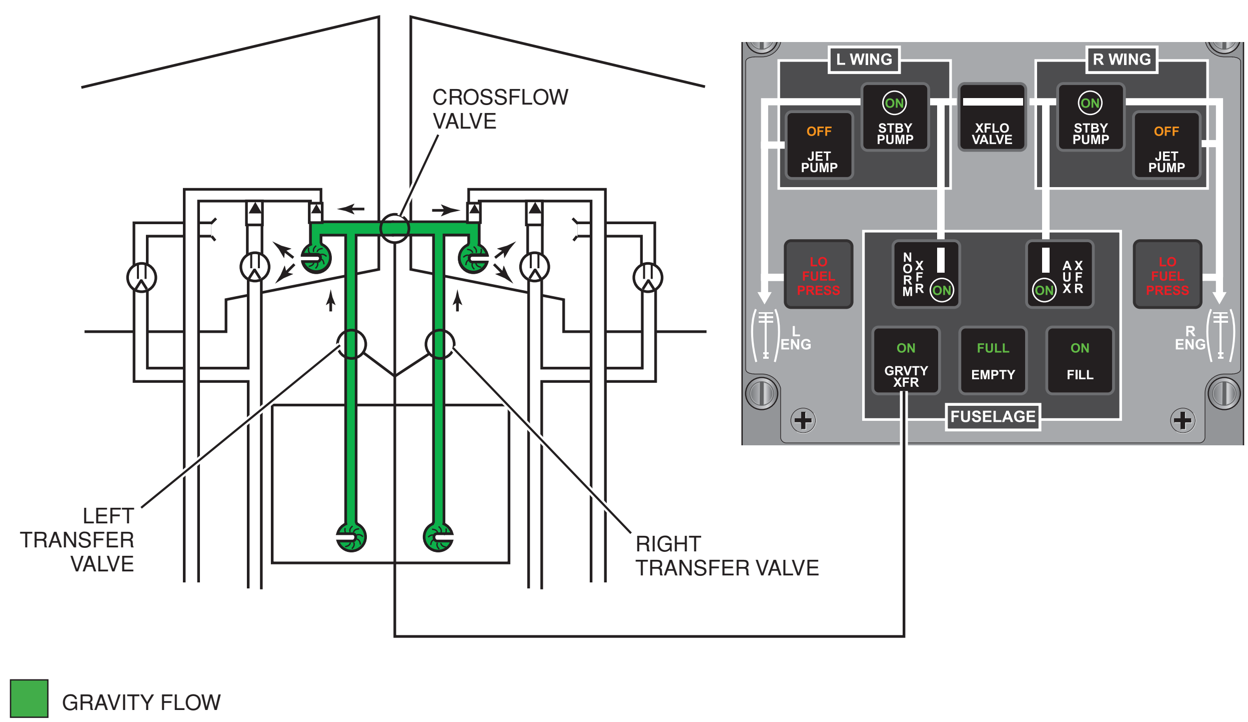


Figure 5-7. Fuselage Tank-to-Wing Tanks (Gravity Transfer)

5 FUEL SYSTEM

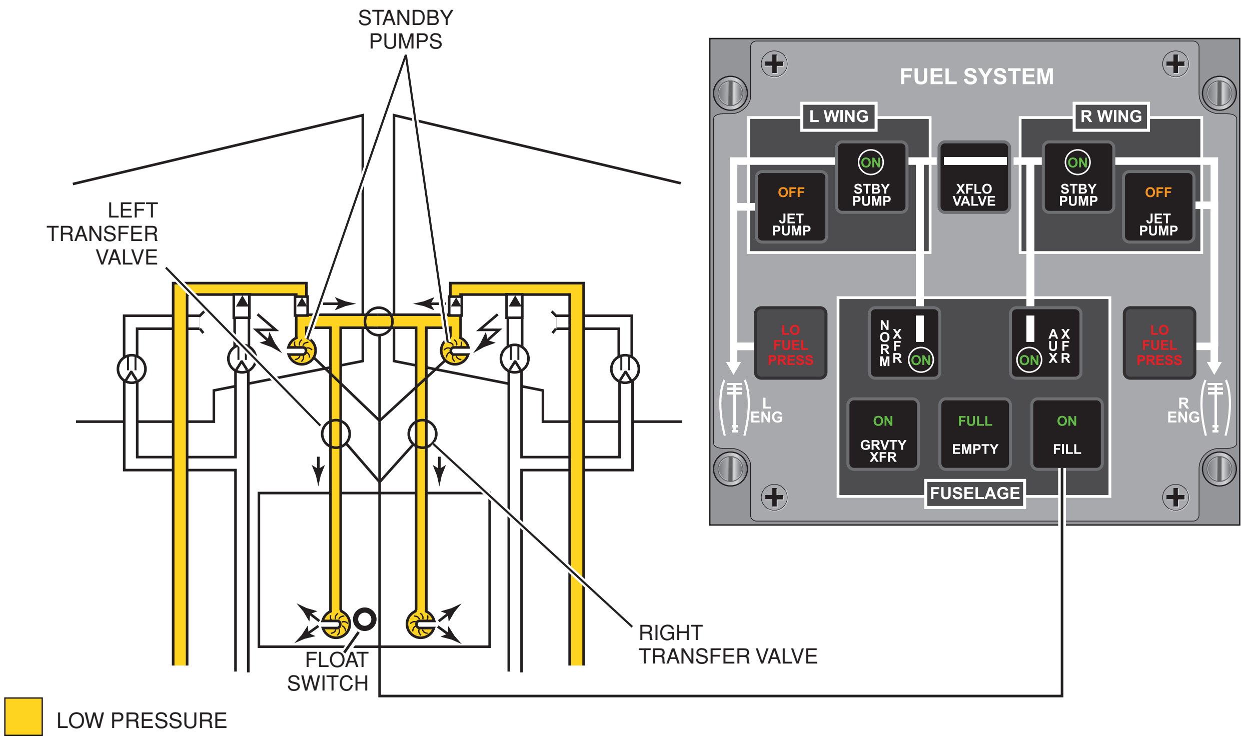


Figure 5-8. Wing Tanks-to-Fuselage Tanks (Fill)

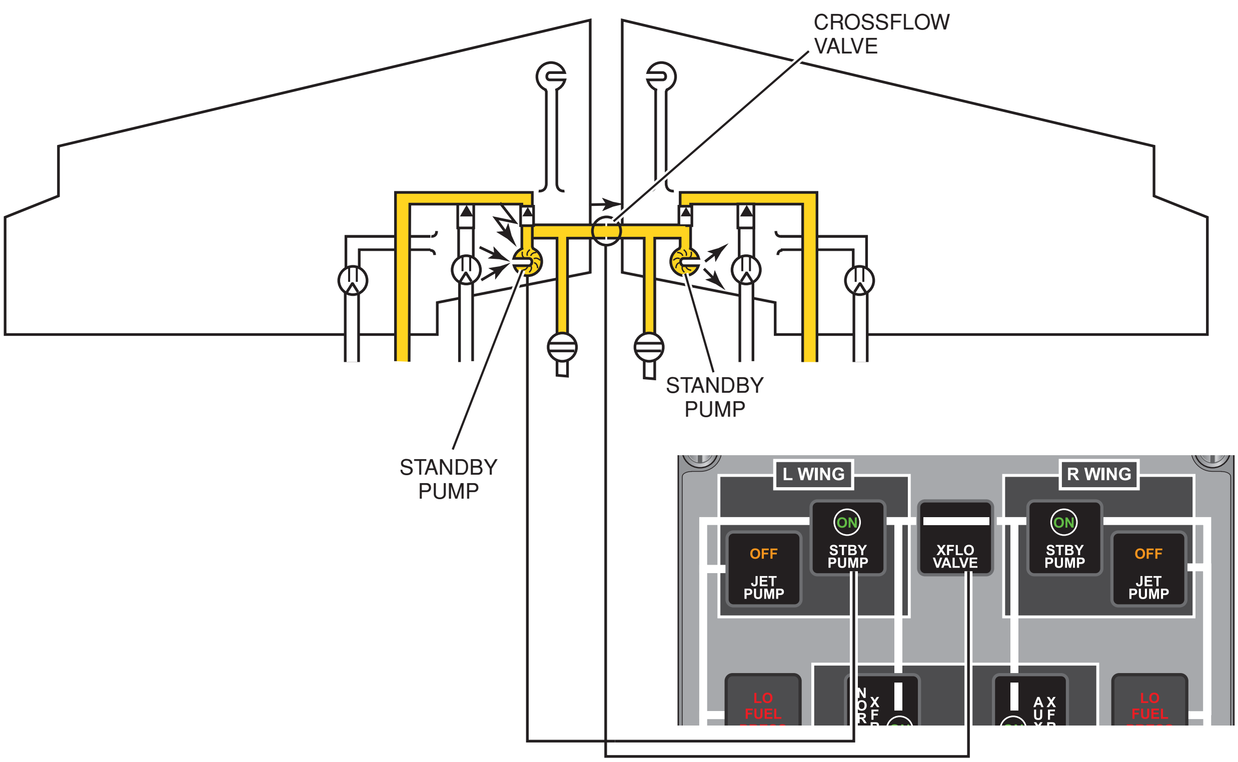


Figure 5-9. Crossflow

5 FUEL SYSTEM

ENGINE FUEL SUPPLY SYSTEM

The transfer jet pumps move fuel within the wings to the area where the main jet pumps and standby pumps are located (Figure 5-10). The transfer jet pumps (three in each wing) operate continuously with the JET PUMP switches ON; they ensure that fuel will be available to the engines in a nose-high attitude when wing fuel quantity is at a low level. The wing (electrical) scavenge pumps operate only when wing fuel quantity is low. If either wings low-level float switch trips (illuminating the LOW FUEL light), only that side's electrical scavenge pump is automatically energized.

The main jet pumps and standby pumps have one-way check valves on the output side to prevent reverse flow when they are inactive.

Jet pumps operate on the venturi principle. They require no electrical power and have no moving parts. Excess high-pressure fuel from the engine-driven pump is routed to the venturi of the jet pump. This high pressure, low-volume stream of fuel draws fuel from the wing tank and provides a low-pressure, high-volume flow from the jet pump.

The jet pumps are controlled with the JET PUMP switches on the fuel control panel which electrically open and close the motive flow control valves. These valves are powered through the respective L and R JET PUMP—XFR VALVE circuit breakers on the pilot/copilot FUEL circuit-breaker panels.

Either the standby pumps or the main jet pumps can supply fuel under pressure to the engine-driven pumps. During engine start, the respective standby pump is automatically energized and the motive flow valve is closed (OFF light on) when the GEN-OFF-START switch is placed in START. The starter disengages at approximately 45 percent N_2 and the motive flow valve automatically opens.

After the engine is started and the START switch is moved from the START position, the standby pump is deenergized and the main jet pump then provides fuel to the engine. If the motive flow valve fails between open and closed, the jet pump OFF light will flash. The standby pumps can be energized at any time to provide fuel pressure to the engines in the event of jet pump failure or if the jet pump switch has been turned off.

FILTERS

The fuel supply for each engine passes through two fuel filters before entering the hydromechanical fuel control unit (HFCU). A secondary airframe filter is located in the supply line between the wing and the engine, and a primary filter is located between the low-pressure stage and the high-pressure stage of the engine-driven fuel pump. Should these filters become clogged, they have a bypass capability. The filters have differential pressure switches which trip whenever the filter is bypassing, or senses an impending bypass, and causes the white L or R ENG FILTERS light on the glareshield annunciator panel to illuminate. The ENG FILTERS lights may illuminate due to a clogged oil filter also. If an ENG FILTER light illuminates in flight, it is not possible to determine which of the filters is clogged, the primary fuel filter or the oil filter. Bypass of a secondary fuel filter will only cause the ENG FILTERS annunciator(s) to illuminate when the aircraft is on the ground. There is an indicator panel in the tailcone, just above the door, which indicates specifically which filter is bypassing.

MAIN FUEL SHUTOFF VALVES (FIREWALL)

The fuel shutoff valves are powered through the L or R FW SOV circuit breakers on the L or R ENGINE group of circuit breakers. The respective L or R FW SOV are controlled by the ENG FIRE PULL T-handles on the glareshield. Pulling either T-handle closes the associated valve, and pushing the T-handle opens the valve provided DC electrical power is available.

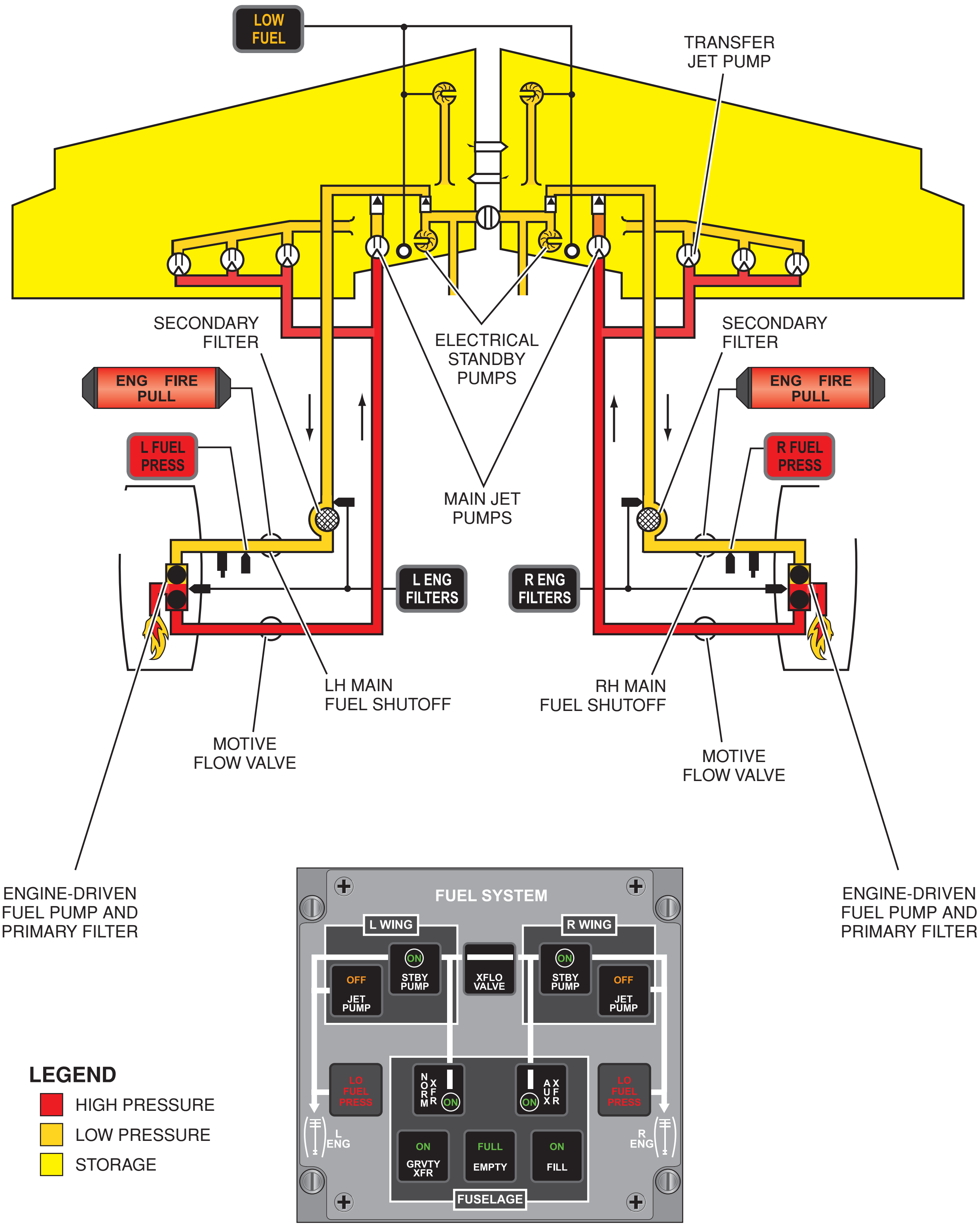


Figure 5-10. Engine Fuel Supply Schematic

FUEL LOW-PRESSURE SWITCHES

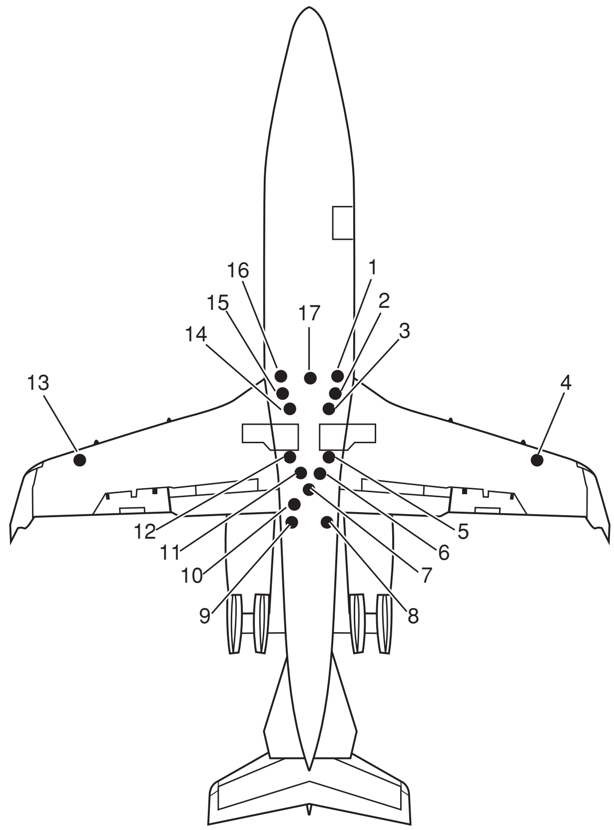
Fuel low-pressure switches, located between the main fuel shutoff valve and the engine-driven fuel pump, sense low pressure in the engine fuel supply line and energize the red L or R FUEL PRESS light on the glareshield annunciator panel and in the fuel panel. When fuel supply pressure drops to 2.75 psi or below, the switch closes. At 3.75 psi, the switch will reopen, turning off the FUEL PRESS light. See *Airplane Flight Manual*, Section III, Emergency Procedures.

PRESSURE-RELIEF VALVE

A high-pressure relief valve is installed in each main fuel line downstream from the main shutoff valves. They relieve pressure buildup caused by thermal expansion of trapped fuel when the engine is shutdown by venting fuel overboard.

RAM-AIR VENT SYSTEM

The ram-air vent system provides pressurization and ventilation for the wing and fuselage fuel tanks in flight (Figure 5-11). There are two ram-air scoops on the underside of each wing. The outboard scoop on each wing ventilates the respective wing tank, and the inboard scoops ventilate the fuselage tank. Ram air enters each outboard ram-air scoop and passes through a wing vent sump and a vent float valve and on into the wing. If the wing is full of fuel, the vent float valve is closed to prevent fuel from flowing overboard through the sump and air scoop. The purpose of the wing vent sump is to catch small amounts of fuel that may seep through the vent float valve when the wing is full. The wing vent sump should be completely drained prior to flight (Figure 5-12). The inboard scoops are connected to the fuselage fuel tank through a fuselage tank vent sump in the bottom of the fuselage.



1. LEFT-WING SCAVENGE PUMP
2. LEFT-WING SUMP
3. LEFT-ENGINE FUEL
4. LEFT-WING VENT (SUMP)
5. LEFT-WING EXPANSION LINE
6. LEFT-WING TRANSFER LINE
7. FUEL VENT (FUSELAGE)
8. LEFT FUEL FILTER
9. RIGHT FUEL FILTER
10. FUSELAGE TANK SUMP
11. RIGHT-WING TRANSFER LINE
12. RIGHT-WING EXPANSION LINE
13. RIGHT-WING VENT (SUMP)
14. RIGHT-ENGINE FUEL
15. RIGHT-WING SUMP
16. RIGHT-WING SCAVENGE PUMP
17. FUEL CROSSOVER

Figure 5-12. Fuel Drains

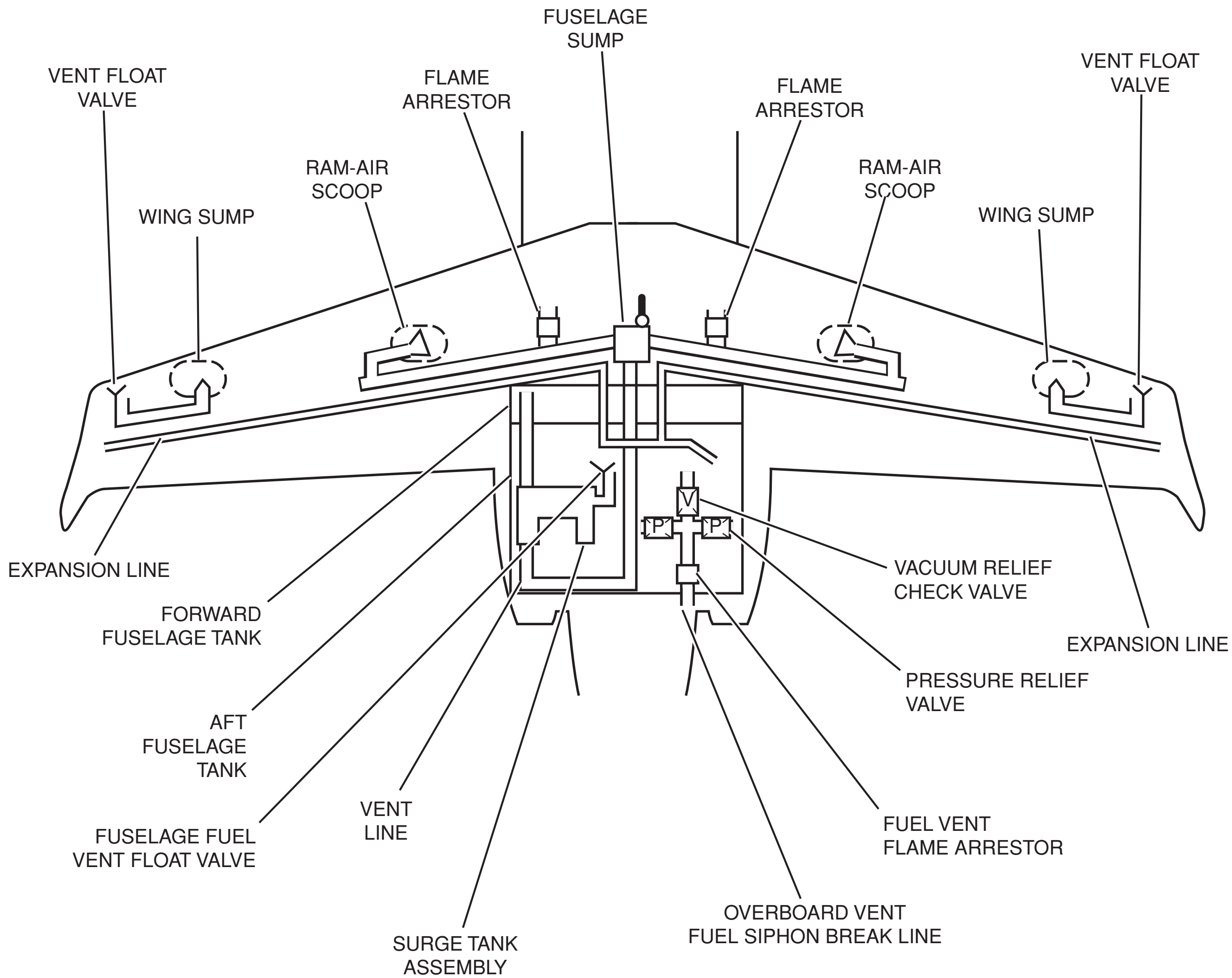


Figure 5-11. Fuel Vent System Schematic

This vent line sump catches fuel that might seep in the vent lines plus moisture that may be taken in through the ram-air scoops. The fuselage vent sump has a push-to-drain tube in the bottom center of the fuselage, slightly aft of the trailing edge of the wings (Figure 5-13). It should be completely drained before flight.

The fuselage tank contains another overboard vent line which is connected to the top aft end of the fuselage tank and extends down and outside the bottom of the fuselage. This vent line contains two pressure relief valves and a vacuum relief check valve, which relieve fuel system pressure buildup due to thermal expansion.

EXPANSION LINES

Two open-ended expansion lines connect the high points outboard in the wing tanks with the high point in the fuselage tank. These lines allow pressure to equalize between all three fuel tanks.

FUEL SYSTEM DRAIN VALVES

Drain valves are located as illustrated in Figure 5-13. Except for the LH and RH wing vent sump drains, all the drain valves are knurled hollow tubes that extend below the skin of the fuselage.

The drain valves are spring loaded in the closed position and sealed. The drains are opened by pushing up on the knurled tubes. The wing vent sump drains are flush with the bottom of the wing and require a screwdriver to open them.

NOTE

Do not twist the drain tubes in the closed position or the O-ring seal may be damaged.

Some of these are moisture drains, but most are fuel system drains located at low points for draining condensation and sediment.

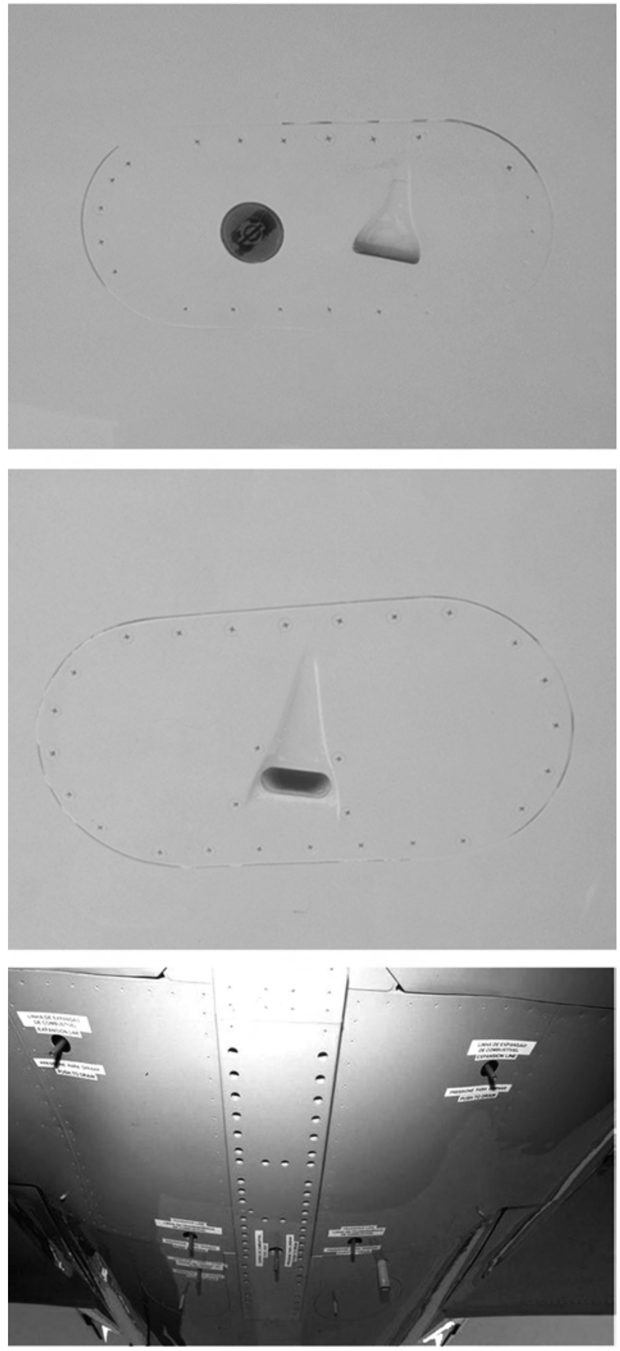


Figure 5-13. Fuel Vent Drain Valves

SINGLE-POINT PRESSURE REFUELING SYSTEM (SPPR)

The single-point pressure refueling system on the Learjet 60 enables wing and fuselage tanks to be filled through a single-point pressure refueling adaptor below the right engine pylon (Figure 5-14).

SPPR CONTROL PANEL

The single-point pressure refueling system control panel is located just forward of the refueling adaptor below the right engine pylon.

The control panel contains all of the switches, valves, and lights necessary to operate the SPPR system.

SPPR Battery Switch

The SPPR BATT switch on the refuel control panel allows operation of the single-point pressure refuel system without the need to enter the cockpit in order to energize aircraft power. When the switch is set to ON, DC power from the right aircraft battery is applied to the SPPR control circuits.

Total-Partial Switch

The control panel contains a two-position switch marked TOTAL and PARTIAL. Selecting the TOTAL position allows all fuel tanks (wing and fuselage) to fill simultaneously. If the PARTIAL position is selected, the wings fill first and then the fuselage tank begins to fill.

The TOTAL position of the refuel selector switch is used to fill the wing and fuselage tanks simultaneously. When TOTAL is selected and refueling pressure is applied (vent valve opens), circuits are completed to open the fuselage tank solenoid valve. When the solenoid valve opens, the fuselage tank shutoff valve opens to admit fuel into the fuselage tank.

The PARTIAL position of the refuel selector switch is used to fill the wings first and then the fuselage. This is useful when full wings and less than full fuselage fuel is desired. When PARTIAL is selected and the vent valve opens, the fuselage tank solenoid valve will be controlled by the wing high-level float switches that complete the circuit to open the fuselage tank solenoid valve. When the solenoid valve opens, the fuselage tank shutoff valve will open and admit fuel to the fuselage tank.

Indicators

The control panel contains two press-to-test indicator lights: a green VENT OPEN light and an amber FUS FULL light.

The green VENT OPEN light, on the refuel control panel, will illuminate whenever the fuselage tank vent valve opens. The light is operated by a microswitch in the valve. The circuit for the fuselage tank solenoid valve is wired through this switch to prevent filling the fuselage tank until the vent valve opens.

The amber FUS FULL light on the refuel control panel will illuminate whenever the fuselage tank float switch actuates. The light illuminates to alert the operator that refuel operations should have automatically terminated. If fuel flow continues with the light illuminated, fueling operations should be immediately terminated.

SPPR COMPONENTS

Refer to Figure 5-14 for location of single-point pressure refueling system components.

Vent Valve

The vent valve is installed to relieve air pressure as fuel fills the tanks. Operation of the valve is checked during the precheck sequence. The valve automatically opens whenever fuel pressure is applied to the system. When the valve reaches the full open position, a switch in the valve completes a circuit to illuminate the VENT OPEN light on the SPPR control panel.

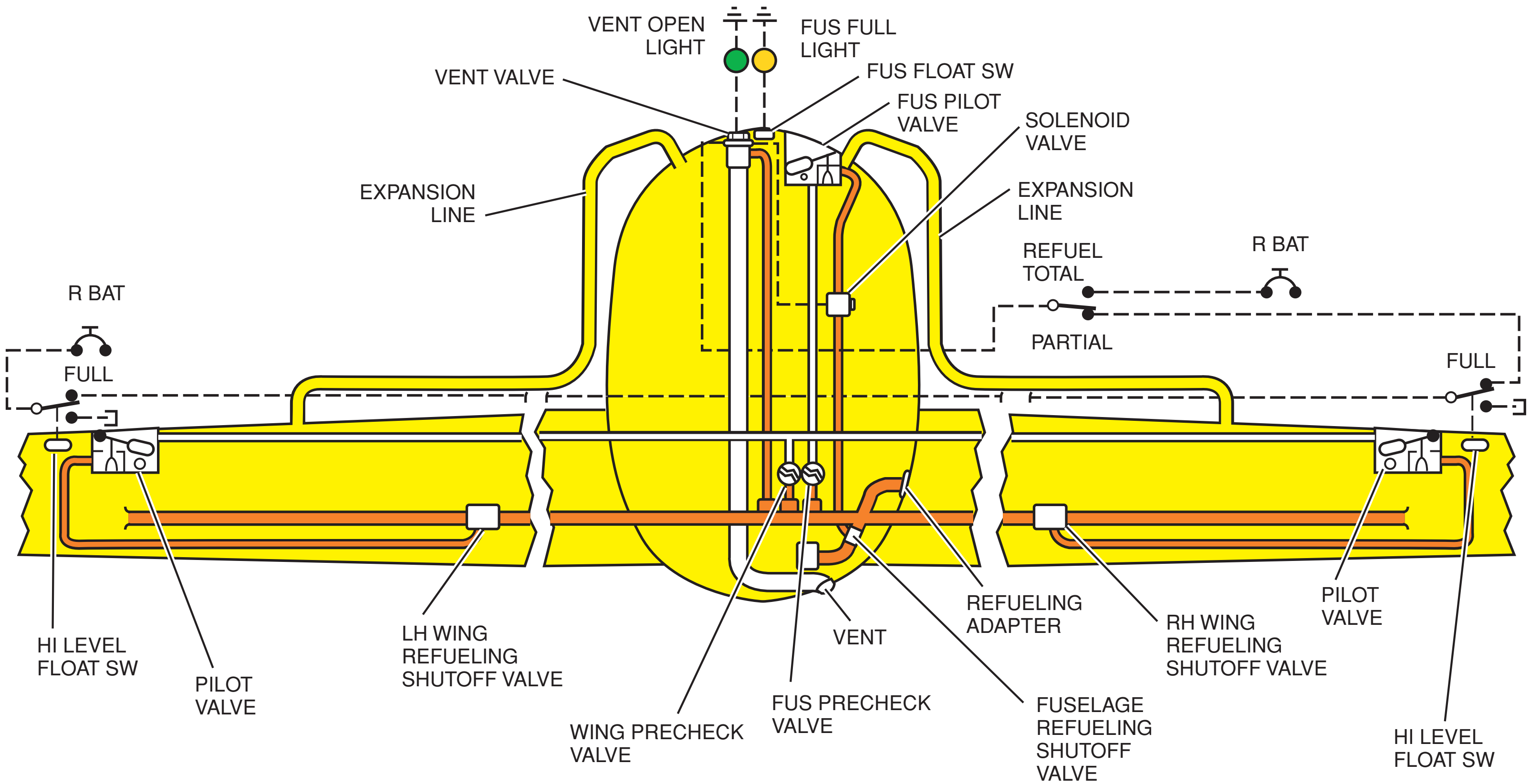


Figure 5-14. Single-Point Pressure Refueling System

Pilot Valves and Shutoff Valves

Each pilot valve, one at the high point in each wing tank and one at the high point in the fuselage tank, controls the respective shutoff valve in each fuel tank. When fuel under pressure enters the refueling adapter, fuel flows to the three refueling shutoff valves. Fuel pressure forces the shutoff valve open. It will remain open as long as fuel pressure on the back side of the shutoff valve poppet is vented through a pilot line to the pilot valve in the corresponding fuel tank. When the fuel tank becomes full, the pilot valve float shuts off vent flow through the pilot line. Fuel pressure then equalizes on both sides of the shutoff poppet, allowing spring pressure to close the shutoff valve poppet.

Solenoid Valve

The pilot line from the fuselage shutoff valve to the fuselage pilot valve contains a solenoid-controlled valve. With the solenoid valve closed, the fuselage shutoff valve cannot be vented through the fuselage tank pilot valve, and the fuselage shutoff valve poppet will close or remain closed. When the solenoid-controlled valve is energized open, the fuselage shutoff valve will be vented through the fuselage pilot valve, thus allowing the fuselage shutoff valve poppet to open. Fuel will flow through the fuselage shutoff valve until the pilot valve closes (that is, the fuselage tank is full) or refueling is terminated. The solenoid-controlled valve is normally closed and can be energized open in two ways: (1) if the TOTAL position on the TOTAL-PARTIAL switch is selected, the solenoid is powered open; and (2) if the PARTIAL position is selected, the valve will be powered open, but only after the high-level float switch in each wing tank trips to the full position. Additionally, in order for the solenoid-controlled valve to be energized open, the fuselage tank vent valve must be fully open. When the vent valve is open (green VENT OPEN light illuminated), power is applied to the solenoid control relay, which closes the circuit to the solenoid-controlled valve.

PRECHECK VALVES

The wing and fuselage precheck valves are used to check operation of the system vent valve and individual shutoff valves before full refueling procedures are commenced. System precheck is accomplished with the refuel selector switch set to TOTAL in order to check all shutoff valves. When the wing and fuselage precheck valves are set to OPEN (grips vertical) and refuel pressure is applied to the refuel adapter, fuel will be admitted to the precheck lines and to the tank fill lines. The shutoff valves will open and fuel will flow into all tanks.

The fuel in the precheck lines will empty into a float basin at each pilot valve. When the basin fills, the pilot valve float will close the pilot valve, which causes the associated shutoff valve to close terminating fuel flow. The vent valve should open when fuel flow is initiated. Fuel flow should stop within 10 to 20 seconds.

REFUELING

See Section II of the *Airplane Flight Manual* and Addenda I, Fuel Servicing, of the *Airplane Flight Manual* for information and servicing procedures.

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QUESTIONS

1. The approximate maximum usable fuel capacities of each fuel tank are:
   A. Each wing—1,424 pounds; fuselage — 3,842 pounds
   B. Each wing—1,449 pounds; fuselage —5,012 pounds
   C. Each wing—1,215 pounds; fuselage — 1,340 pounds
   D. Each wing—1,160 pounds; fuselage —2,603 pounds

2. Which pair of fuel pumps may be checked prior to engine start by depressing the TEST switches on the glareshield annunciator panel and listening for an audible indication?
   A. Wing Scavenge pumps
   B. Fuselage pumps
   C. Standby pumps
   D. Transfer pumps

3. The switch which controls the RH transfer system is:
   A. FUS TANK GRVTY XFR
   B. FUS TANK NORM XFR
   C. FUS TANK AUX XFR
   D. CROSSFLOW

4. The standby pumps are used for all except one of the following functions:
   A. Engine start
   B. As backup for main jet pumps
   C. Wing-to-wing crossflow with a main jet pump inoperative
   D. Wing-to-fuselage transfer of fuel

5. The green FUEL SYS light will flash when:
   A. Fuselage tank is full and FILL is selected.
   B. Fuselage tank is empty and NORM XFR, AUX XFR or RAPID XFR is selected.
   C. A low-fuel pressure indication is present at the engine.
   D. Both A and B are correct.

6. Using procedures in Section II of the *Airplane Flight Manual*, NORM XFR may be initiated:
   A. At any time regardless of wing fuel quantity
   B. When the fuel level in the wing tanks indicates 1,200 pounds or less
   C. When the WING FULL lights extinguish
   D. Only when the fuel level in each wing has decreased to 410 pounds

7. Rapid fuselage fuel transfer to the wings is accomplished:
   A. Through both transfer lines with the fuselage tank FILL switch set to ON
   B. Through both transfer lines with the fuselage tank GRVTY XFR set to ON
   C. Through the right transfer line with the fuselage tank AUX XFR switch set to ON
   D. Through both transfer lines with both the fuselage tank NORM XFR and AUX XFR switches selected to ON

8. The amber LOW FUEL light illuminates when:
   A. 350 pounds total fuel remain
   B. Approximately 410 pounds of fuel remain in either wing
   C. 400 to 500 pounds total fuel remain
   D. More than 500 pounds of fuel remain in each wing

9. Which pumps automatically energize when the LOW FUEL light illuminates?
   A. Standby pumps
   B. Main jet pumps
   C. Transfer jet pumps
   D. Wing scavenge pumps

10. The crossflow valve opens:

A. Only when the XFLO VALVE switch is set to open
B. Only when the XFLO VALVE switch is set to open or the NORM XFR switch is set to ON
C. Anytime electrical power is lost
D. Whenever XFLO VALVE switch is set to open or the NORM XFR, AUX XFR, GRVTY XFR or FILL switches are set to ON

11. The crossflow valve should not be opened:

A. When a main jet pump is inoperative, unless using rapid transfer of fuselage fuel to the wings
B. When either engine fails
C. During gravity transfer
D. When a standby pump is inoperative

12. A steady green FUEL SYS light on the annunciator panel indicates:

A. Any transfer/fill operation is in progress
B. XFLO valve is selected open
C. A STBY PUMP is selected ON
D. All of the above

13. Approximately how much fuel will not transfer to the wings using gravity transfer?

A. 1,000 pounds
B. 500 pounds
C. 350 pounds
D. None

14. Which of the following fuselage tank transfer switches will cause the standby pumps to be disabled?

A. GRVTY XFR switch ON
B. NORM XFR switch ON
C. AUX XFR switch ON
D. Any of the above

15. Which transfer method provides for automatic closing of a transfer valve when one, or both, wing tanks become full?

A. Gravity transfer
B. Normal transfer
C. Auxiliary transfer
D. None

16. Which transfer method provides for automatic closing of the transfer valve(s) and pump shutdown when the fuselage tank is empty?

A. Gravity transfer
B. Normal transfer
C. Auxiliary transfer
D. None

17. The white L or R ENG FILTERS light, on the ground, indicates:

A. An impending bypass of the aircraft fuel filter
B. An impending bypass of the engine fuel filter
C. An impending bypass of the engine oil filter
D. Any of the above

18. Illumination of the red L or R FUEL PRESS light indicates:

A. Fuel pressure to the respective engine fuel pump is low
B. Fuel pressure to the respective engine is too high for safe operation
C. An L or R engine fuel filter is bypassing
D. Fuel pressure to the respective engine is optimum for engine start

CHAPTER 6
AUXILIARY POWER UNIT

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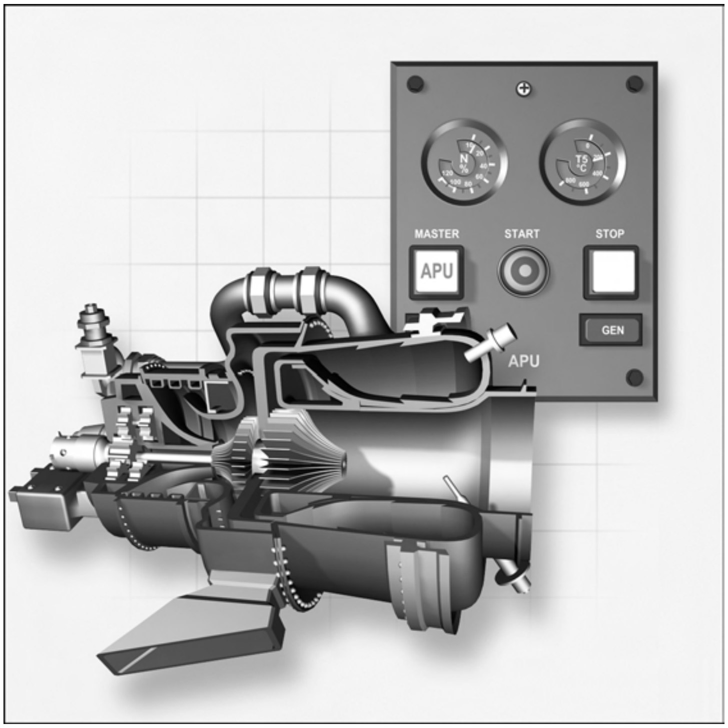
CHAPTER 6
AUXILIARY POWER UNIT



INTRODUCTION

The auxiliary power unit (APU) system on the Learjet 60 aircraft is optional equipment and designed for ground use only. The APU supplies electrical power to the aircraft electrical system for battery charging, engine starting and ground operations. The APU cannot be utilized in conjunction with a ground power unit (GPU), but a GPU may be used to start the APU.

GENERAL

The optional APU engine installed in the Learjet 60 aircraft is a self-contained gas turbine manufactured by Sundstrand-Turbomach, model number T-20G-10C3. The APU system is installed in the Learjet 60 under a Supplemental Type Certificate (STC) held by PATS, Inc.

The APU is located in the rear equipment bay, aft of the tailcone baggage compartment (Figure 6-1). It is enclosed in a fireproof containment box equipped with its own fire detection and extinguishing system. The APU is limited to ground operation only, and will automatically shutdown if left on for takeoff.

Figure 6-1. APU Access Door

The primary function of the APU is to supply 28-VDC electrical power to the aircraft electrical system through the APU generator.

The APU is independent of all aircraft systems except for the DC power supply, used for energizing the fuel system and the starter/generator.

Figure 6-2. APU Air Intake/Exhaust Port

MAJOR COMPONENTS

The APU consists of six major sections:

- Air intake
- Compressor
- Combustor
- Turbine
- Exhaust
- Accessory gearbox

AIR INTAKE

The air intake is located on the lower right side of the rear fuselage (Figure 6-2). Airflow is drawn in through the air intake and is directed to the upper and lower inlet duct halves. Inlet screens are attached to the housing and provide protection against foreign object ingestion. See "Section I" of the Learjet 60 *Pilot's Manual* for danger areas around the APU intake and exhaust.

COMPRESSOR

The compressor is a single-stage centrifugal-type that draws air in through the air intake and compresses it for combustion.

COMBUSTOR

The combustor section consists of a single, annular, combustion chamber, igniter plug, and fuel nozzles inserted into the turbine housing. Within the combustor, the fuel air mixture is ignited to provide the energy that drives the turbine and compressor assembly.

TURBINE

A single-stage radial turbine is mounted through a shaft to the compressor. The turbine is designed to extract almost all of the energy from the expanding gases. Most of the energy is used to drive the compressor, the gearbox, and APU accessories.

EXHAUST

The exhaust assembly attaches to the APU and directs the exhaust gases into the atmosphere. The exhaust outlet is located on the right upper side of the aft fuselage (Figure 6-2). The exhaust assembly, through an eductor method, also provides cooling for the starter/generator and other components within the APU enclosure.

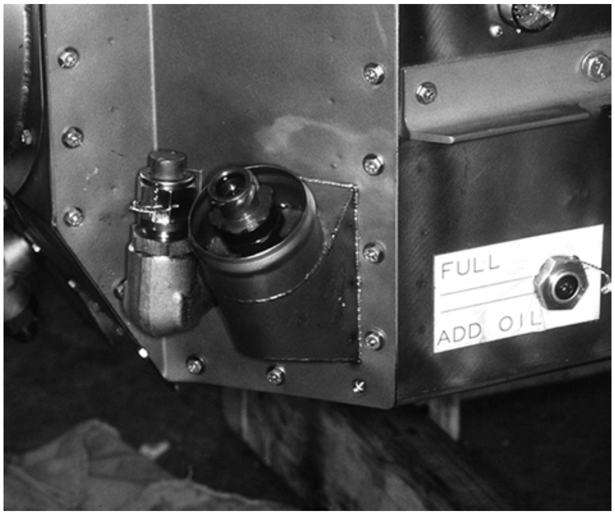


Figure 6-3. APU Oil Filler Cap and Sight Glass

ACCESSORY GEARBOX

An accessory gearbox forms an integral part of the APU. The gearbox is designed to convert the high turbine speed to the values required to drive the APU generator and accessories. The accessories include the lubricating pump, fuel control and fuel pump unit, and the starter/generator.

APU SYSTEMS

OIL SYSTEM

The oil system is a self-contained, two-quart capacity, fully automatic system which provides lubrication for the bearings and for the accessory gearbox. Checking APU oil level (Figure 6-3) through the sight glass is performed during the aircraft exterior inspection.

FUEL SYSTEM

The APU fuel system is fully automatic with no operator controls or adjustments required. Fuel is supplied to the APU from the aircraft's left wing fuel tank (60-073 and subsequent) and flows through a shutoff valve (60-001through 60-072 have fuel supplied from the fuselage tank, see *AFM*), a boost pump, and a fuel filter. Before an APU start, the operator must ensure that there is a minimum of 100 pounds of fuel in the left wing or fuselage tank for APU operation.

When the APU control panel MASTER push-button (Figure 6-4) is depressed, it opens the shutoff valve and energizes the boost pump. The boost pump then provides a fuel flow to the APU fuel system.

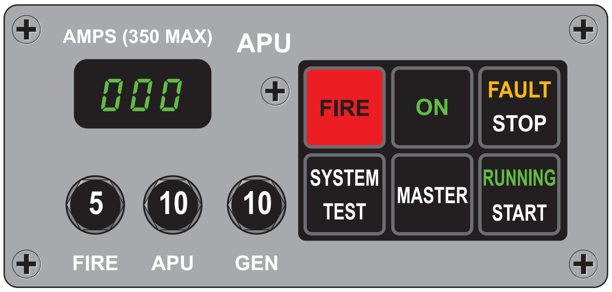


Figure 6-4. APU Control Panel

If the valve fails to close on shutdown, the APU FAULT light and the master caution lights will illuminate, if not previously inhibited.

During normal operation, the fuel control unit (FCU) will schedule fuel flow to the combustion chamber in order to maintain rpm at 100%. This fuel scheduling allows the power developed by the APU to be equal to the power

required and thereby maintain a constant rpm throughout all APU load variations.

NOTE

The fuel consumption rate of the APU is approximately 40 lbs per hour and is affected by pressure altitude, outside air temperature, and the load imposed; therefore, fuel consumption at a given pressure altitude varies.

IGNITION SYSTEM

A fully automatic ignition system provides high voltage electricity to the single igniter plug, located within the combustion chamber. The ignition system is controlled by the APU relay box assembly.

EXHAUST GAS TEMPERATURE (EGT) SYSTEM

APU exhaust gas temperature is sensed by a single thermocouple located in the APU exhaust. The thermocouple provides exhaust gas temperature indications to the electronic sequence unit (ESU). The ESU continually monitors the thermocouple circuit whenever APU electrical power is supplied.

ELECTRICAL CIRCUIT PROTECTION SYSTEM

All APU circuit breakers are 28 VDC. Three are located inside the cockpit on the APU control panel, just above the copilot's circuit-breaker panel (Figure 6-5), and three additional circuit breakers are located within the APU aft compartment on the APU relay box assembly (Figure 6-6).

FIRE WARNING AND EXTINGUISHING SYSTEM

The fire bottle is located in the tail area adjacent to the APU. It is equipped with a pressure switch, tied to the APU fault indication system, to prevent starting the APU with low fire bottle pressure.

Figure 6-5. APU Control Panel Location

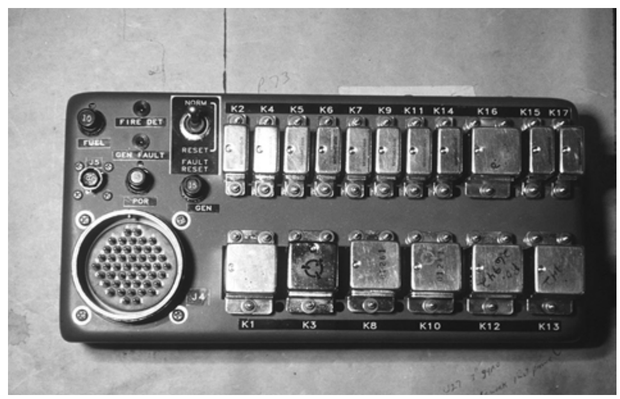


Figure 6-6. APU Relay Box Assembly

The fire loop is installed around the APU engine and the DC generator, inside the stainless steel fire containment box (Figure 6-7). The fire loop will activate at 800°F at a single point, or 375°F within the overall length, to close the APU fuel shutoff valve and discharge the Halon extinguishing agent into the containment box. This action will also shut down the APU, sound the fire warning alarm, located in the left wheel well (Figure 6-8), and illuminate the APU FIRE warning light pushbutton (see Figure 6-4).

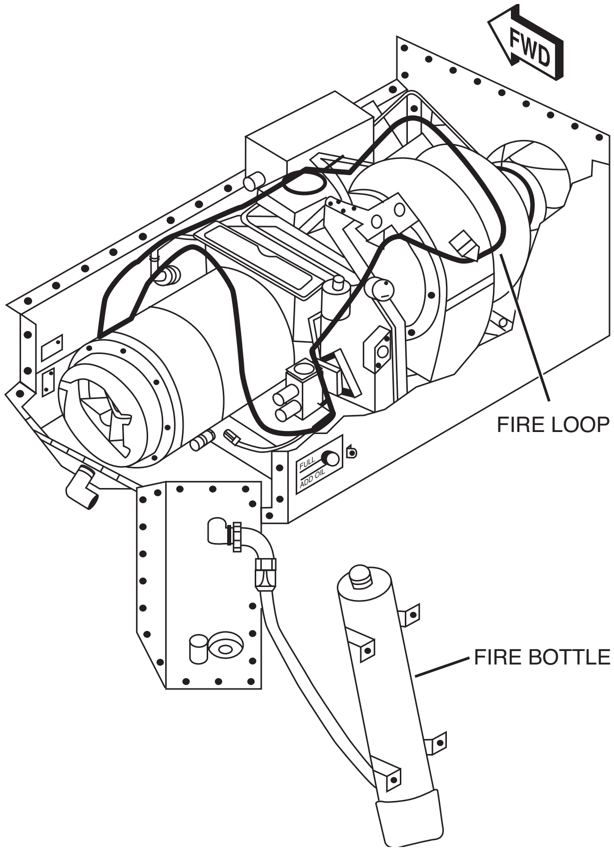


Figure 6-7. APU Fire Detection

CAUTION

Illumination of an amber or red APU control panel annunciator will also illuminate the corresponding master warning/caution lights on the main instrument panel. Care must be taken to insure that the APU control panel is checked if the master warning or caution lights illuminate and there are no annunciators illuminated on the main annunciator panel that would have activated them.

The fire extinguisher pressure and the integrity of the fire loop is tested each time, before the APU is started, with the APU SYSTEM TEST pushbutton. If the pressure is low or the loop is shorted, the APU fuel valve will not open and the APU cannot be started. Additionally, an OVERHEAT "bite" on the APU relay box

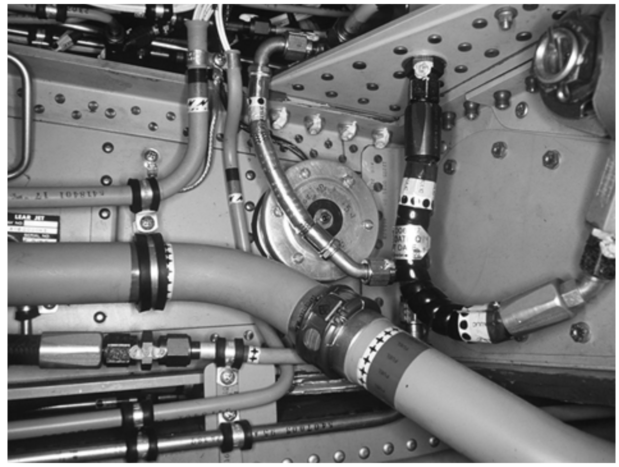


Figure 6-8. APU Fire Warning Bell Location

assembly (also located in the rear equipment bay) will indicate if the fire loop has sensed an overheat or fire condition.

CONTROL/INDICATION SYSTEM

The control/indication system requires a minimum amount of control inputs and warning outputs for proper operation. The APU engine/generator-starter combination is one integral operating unit. There is no separate APU generator on/off switch. Any APU generator fault will shutdown the APU engine.

APU, APU generator and overheat BITE malfunction indicators are located within the APU compartment. Also located inside the APU compartment is the FAULT indication RESET switch (refer to Figure 6-10) on the relay box assembly.

The APU is controlled from the APU control panel, located above the copilot circuit-breaker panel (see Figure 6-4), and includes the following pushbuttons and lights:

This pushbutton supplies power to the APU system. The legend is daylight readable and is illuminated white when the aircraft NAV light switch is on.

This light is illuminated green when the MASTER switch is on and supplying power to the APU system.

This pushbutton is used to start the APU system. The bottom half, labeled START, is illuminated white whenever the MASTER pushbutton is on, to identify the switch.

Depressing the START pushbutton initiates the APU start sequence. The top half of this button is labeled RUNNING and is illuminated "green" whenever the APU is operational and supplying electrical power to the aircraft.

If the APU system is operating with ground power (GPU) attached to the aircraft, the APU generator will be kept off-line. The RUNNING half of the switch will be illuminated, indicating that the APU system is ready to supply power to the aircraft. However, the AMP meter (see Figure 6-4) will display 000, showing that the generator is off-line.

This is the only time that the APU system will display a RUNNING indication without the APU actually supplying power to the aircraft. Normally, when the RUNNING light is illuminated, some current draw will be displayed on the AMP meter.

This pushbutton is used to stop the APU system and to indicate a fault. The bottom half, STOP, is illuminated white whenever the master switch is on to indicate switch function. During normal operation, this pushbutton is used to shutdown the APU by applying an overspeed signal to the electronic sequence unit (ESU) of the APU.

A normal shutdown operation will not cause the FAULT half of the pushbutton to illuminate. Any APU system fault, whether it be an engine or generator problem, will cause the FAULT and MASTER CAUTION light to illuminate, as well as cause an APU shutdown.

The Fault indicator is illuminated amber by the malfunction signal or by the fuel valve position-sensing circuit. The APU malfunction (MALF) signal is output from the ESU whenever the APU shuts down.

The STOP pushbutton simulates an overspeed input to the ESU; however, the FAULT indicator will not illuminate after the STOP pushbutton is pressed. The STOP pushbutton will not clear any preexisting faults.

The FAULT indicator circuit is latched and can be cleared by the FAULT RESET switch on the APU relay box assembly, located forward of APU containment box (Figures 6-4 and 6-6). The circuit does not latch when the fault is caused by the squat switch circuit on takeoff. This fault can be cleared by selecting the APU MASTER pushbutton to the OFF position.

The FAULT indicator is also illuminated whenever the fuel valve is not closed and the MASTER pushbutton is off. This cannot be cleared by the FAULT RESET pushbutton. The valve must be visually inspected to verify it is closed prior to flight.

This guarded switch is an APU system fire indicator and extinguishing pushbutton.Should there be a fire in the APU, as detected by the integral fire loop, the FIRE pushbutton will illuminate red, and the aircraft's master warning lights will illuminate. Detection of an APU fire will automatically sound the APU fire alarm, close the APU fuel valve as well as start a timing sequence in the fire-extinguisher system.

If the FIRE pushbutton is ***not*** depressed manually, the fire extinguisher will be activated after an approximate 15-second time delay, following illumination. Raising the guard and depressing the FIRE pushbutton will immediately activate the fire-extinguisher system.

This pushbutton tests the integrity of the fire detection and extinguishing system of the APU. A successful test will illuminate the FIRE pushbutton, the aircraft's master warning and caution lights, sound the APU fire alarm and close the APU fuel shutoff valve.

Activation of the fire-extinguisher time delay is inhibited during a system test. The legend is daylight readable and illuminated "white" when the aircraft NAV light switch is on.

Operation of the SYSTEM TEST pushbutton while the APU is operating will cause the APU to shut down in approximately 15 to 20 seconds.

This switch also activates a test of all annunciator lamps on the APU control panel. This allows a complete test prior to starting the APU. Since the system test switch illuminates amber, during the test, this will also illuminate the master caution lights if not previously inhibited.

The AMP meter cannot be tested with the cockpit annunciator light test unless the APU MASTER switch is on.

ELECTRONIC SEQUENCE UNIT (ESU)

The electronic sequence unit (ESU) automatically controls the APU start sequence and monitors engine speed, temperature and oil pressure. The ESU contains circuitry which will shut the engine down if any of the following parameters are exceeded:

AUTOMATIC SHUTDOWN FEATURES

- Overspeed................................. > 106%
- Underspeed.............................. < 90%
- EGT overtemp........... 1,300°F (704°C) (max duration 0.5 seconds)
- Low oil pressure............. < 6 psig ±1
- High oil temperature.............. > 275°F
- Loss of EGT signal(open thermocouple circuit)
- Loss of rpm (open motional pickup transducer circuit)
- ESU overcurrent

Each time the APU STOP switch is activated it feeds an overspeed signal to the ESU which shuts down the APU and tests the overspeed circuitry.

The ESU has a separate bite box, located in the APU compartment (Figure 6-9), and contains five (5) BITE indicators to assist in troubleshooting for maintenance purposes.

Figure 6-9. APU Bite Indicator Box

NOTE

A normal shutdown will indicate an overspeed on the bite indicator box.

For Bite Indicator Box light code analysis and troubleshooting refer to the *Sundstrand Maintenance Manual*.

ACCESSORY SYSTEMS

APU RELAY BOX ASSEMBLY

The APU relay box assembly, in the tailcone, contains two (2) BITE indicators to display generator and overheat faults (Figure 6-10).

If the generator trips off-line due to a generator fault, the APU will shut down by simulated overspeed and the GEN FAULT indicator will change to white. This can be cleared by moving the spring loaded FAULT RESET switch to RESET momentarily.

Figure 6-10. APU Relay Box (Bite Indicators and Fault Reset)

If the fire detector senses a fire or compartment overheat, the FIRE DET bite will change to white. This can also be cleared by the FAULT RESET switch in the same manner.

AIRCRAFT SQUAT SWITCH

Although it is not an operator input, the aircraft squat switch (Figure 6-11) is an input to the APU system. The APU is approved for ground operation only. The squat switch will ensure that if the APU is mistakenly left operating for takeoff, an automatic shutdown will occur. A FAULT indication on the APU control panel illuminates after takeoff and the master caution light illuminates on the pilot and copilot instrument panel. The APU MASTER switch must be switched to OFF to clear the FAULT indication.

Figure 6-11. Aircraft Squat Switch

HOURMETER

An hourmeter (Figure 6-12) is installed on the APU to record the total operating time of the unit. Maintenance inspections will be scheduled based upon elapsed operating times. The hourmeter is operational whenever the APU is in operation.

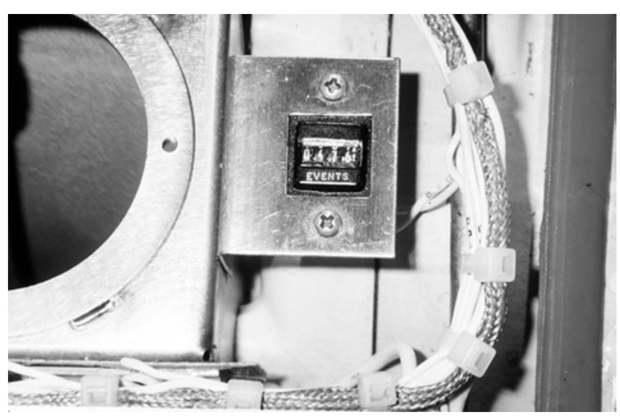


Figure 6-12. APU Hourmeter

LIMITATIONS AND GENERAL/OPERATIONAL LIMITS

GENERAL

Certification Status

The aircraft is certified in accordance with 14 CFR Part 25.

Aircraft Flight Manual Limitations

The limitations in Section I of the *Aircraft Flight Manual* are applicable with the addition of the following:

Type Of Operation

This auxiliary power unit is approved for:

GROUND OPERATION ONLY

OPERATIONAL LIMITS

- APU must NOT be operated unattended with passengers on board.
- APU must be attended while operated during pressure refueling.

CAUTION

APU must not be operated during aircraft fluid deicing.

Ambient Temperature Limitations

–43°C	to	+54°C
–45°F	to	130°F

APU Starter Limits

START	MAX TIME ON	FOLLOWED BY
1	30 Sec ON	2 Min OFF
2	30 Sec ON	2 Min OFF
3	30 Sec ON	30 Min OFF

After 3rd attempt, wait 30 minutes.

APU Generator Limits

Continuous Operation—350 amps maximum

Transients—Higher transient loads for cross-starts and battery charging, up to maximum generated output, are authorized. The indicator on the APU control panel will flash at or above 400 amps.

Approved Fuels

In accordance with MIL-T-5624 JP5 or ASTM D1655 (Jet A, Jet A-1 or kerosene-type fuels).

Approved Oils

Refer to Table 6-1 (In accordance with MIL-L-7808 or MIL-L-23699)

Table 6-1. APPROVED APU OILS

Oil Temperature -54°C to 54°C
Aeroshell Turbine Oil 390
Brayco 880
British Petroleum Aero Turbine Oil 15
Castrol 3C and Castrol 325
Exxon/Esso Turbo 2389
Oil Temperature -40°C to 54°C
Aeroshell 560
Castrol 98 and Castrol 5000
Exxon/Esso Turbo Oil 2380 and Exxon ETO 85
Hatcol 3611
Mobile Jet Oil II and Mobile Jet Oil 254
Royco/Aeroshell Turbine Oil 500/555
Royco 899

NOTE

Do not mix different brands of oil.

APU/AIRCRAFT ELECTRICAL SYSTEM INTERFACE

The APU generator will automatically supply 28- VDC electrical power to the aircraft electrical system and battery charging bus (Figure 6-13) after the APU is started and the RUNNING annunciator is illuminated. However, if a GPU is supplying electrical power to the aircraft, the APU generator will be kept off-line. The APU will share the aircraft electrical load with either or both aircraft engine mounted generators if they are on. The electrical load distribution can be observed on the electrical power monitor (Figure 6-14) and the AMPS indicator on the APU control panel (see Figure 6-4).

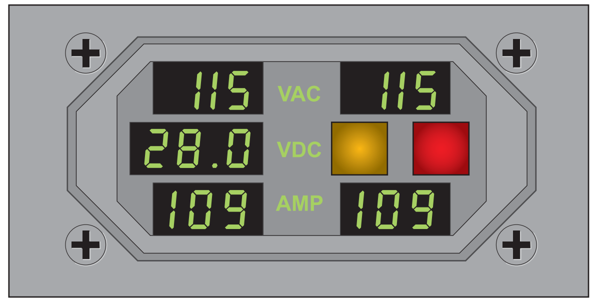


Figure 6-14. Aircraft Electrical Power Monitor

NORMAL OPERATION

The normal procedures in Section II of the *Airplane Flight Manual* are applicable with the addition of the following:

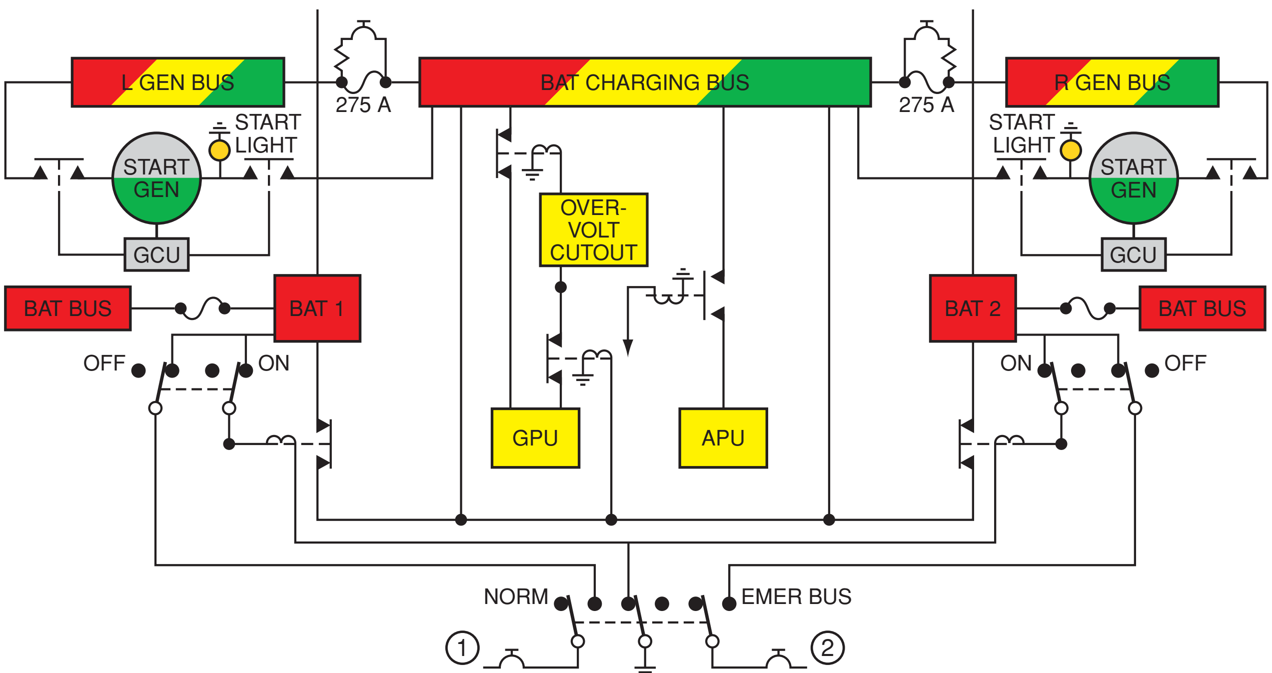


Figure 6-13. APU/Aircraft Electrical Interface

APU PRESTART INSPECTION

Exterior Inspection

NOTE

A flashlight may be required for performing these checks.

Tailcone Interior

Check for the following:

1. APU oil level—Add oil if below ADD line

WARNING

The following precautions must be observed when handling turbine oils:

Prolonged contact with lubricating oil meeting MIL-L-7808 may cause skin rash. The areas of skin and clothing contacting this oil should be thoroughly washed immediately. Saturated clothing shall be removed immediately. Areas in which this oil is used shall be adequately ventilated to reduce mist and fumes to a minimum.

Frequent skin contact with lubricating oil meeting MIL-L-23699 may result in permanent paralysis since this oil contains an additive that is poisonous and readily absorbed through the skin. Do not allow this oil to remain on the skin longer than necessary.

2. Flammable fluid leaks on all aft fuselage components
3. Circuit breakers—three on the APU relay box assembly IN
4. APU inlet and exhaust clear

Interior Inspection

1. CIRCUIT BREAKERS (3) IN (APU control panel)
2. BATTERY 1, BATTERY 2 switches selected ON
3. GPU (if desired)—connect
4. Verify that 18 volts minimum are available for starting
5. Left wing fuel quantity—Check

NOTE

The APU fuel is drawn from the fuselage or left wing fuel tank. Minimum left wing or fuselage fuel quantity for APU operation is 100 pounds.

6. Press APU MASTER switch. Verify—ON, START, STOP and AMP meter all illuminate.

NOTE

30 seconds after the APU MASTER switch is depressed, the APU FAULT light illuminates, if an APU start has not been initiated.

7. APU SYSTEM TEST switch—Press. Check that the APU FIRE warning switch illuminates, that the aircraft fire alarm sounds, and all APU annunciator lights illuminate. The digital AMP meter will display all 8s.

NOTE

The APU SYSTEM TEST switch also tests the integrity of the fire loop and fire-extinguisher pressure. If the fire-extinguisher pressure is below 375 psi or the fire loop is inoperative the APU cannot be started.

APU START

1. BCN/STROBE light switch—BCN

WARNING

Check that the exhaust area is clear of personnel and equipment prior to APU start.

2. APU START switch—Press momentarily (approximately three seconds). The APU start sequence is automatic but should be monitored to verify that the following events occur:

- The RUNNING light illuminates to indicate the APU is operational and the generator is able to supply power.
- The generator will automatically come on line if a GPU is not used.
- Monitor A/C and APU ammeters.

NOTE

The MASTER WARNING/CAUTION light may illuminate during APU start.

3. GPU (if applicable)—Disconnect

APU SHUTDOWN

1. STOP switch—Depress. The RUNNING light will extinguish and the APU generator will go off-line.

NOTE

The STOP switch is not effective until the APU has achieved 90% rpm. The APU MASTER switch must be switched to OFF to abort a start.

2. Press the APU MASTER switch OFF. The ON light will extinguish.

EMERGENCY PROCEDURES

The Emergency Procedures in Section III of the *AFM* are applicable with the addition of the following:

APU FIRE

If the APU FIRE warning light illuminates and the fire alarm sounds indicating an APU fire:

1. STOP switch—Depress. (If the APU has not already shut down automatically)
2. APU MASTER switch—OFF
3. APU FIRE switch (lift cover)—Depress

NOTE

The APU FIRE warning light remains illuminated until the fire is extinguished.

4. If the APU FIRE light remains illuminated, follow Emergency Evacuation procedures as outlined in the basic aircraft flight manual.

ABNORMAL PROCEDURES

The Abnormal Procedures in Section IV of the *AFM* are applicable with the addition of the following:

APU FAULT LIGHT ILLUMINATED

If a fault indication is displayed on the APU control panel, the master caution lights are illuminated and the APU shuts down. Anytime a fault light has illuminated, the APU compartment should be visually inspected for general condition prior to flight to ensure there is no visible damage to the APU containment box.

WARNING

Do not dispatch with a FAULT indicator illuminated.

NOTE

Other than a squat switch controlled shutdown on takeoff, or a failed fuel valve, an APU FAULT indication should only be cleared by actuating the APU FAULT RESET switch on the APU relay box in the APU compartment. Cycling the aircraft batteries will clear some APU fault indications, however this procedure should not be used to clear APU faults.

The APU will automatically shut down for the following reasons:

1. Overspeed—Engine speed of over 106% rpm
2. Underspeed—Engine speed of under 90% rpm
3. Overtemperature—EGT 1,300° F with a maximum duration of 0.5 seconds
4. Loss of the EGT signal to the APU ESU
5. Low oil pressure (< 6 psig ±1)
6. High oil temperature (> 275° F)
7. Loss of rpm
8. APU fire indication
9. Low fire-extinguisher bottle pressure (< 375 psi)
10. Generator malfunction

Following FAULT light illumination:

1. APU MASTER pushbutton—Depress to turn the MASTER off. The FAULT indicator will remain illuminated.
2. Visually inspect APU and APU compartment prior to flight to ensure that the APU containment box has not been damaged. If visibly damaged, obtain maintenance assistance to determine cause and corrective action.
3. APU FAULT BITE indicators (located on the bite indicator box, in the APU compartment)—Record BITE readings to assist maintenance troubleshooting prior to clearing the faults by resetting the FAULT switch.
4. APU FAULT RESET switch on the APU relay box assembly—Reset
5. If FAULT indicator remains illuminated, the APU fuel valve is not closed. Obtain maintenance assistance to determine cause and corrective action.

INTENTIONALLY LEFT BLANK

QUESTIONS

1. The APU will automatically shut down, if left on for takeoff, when:
- A. The aircraft's groundspeed is above 60 knots
- B. The aircraft squat switch sends a signal to the electronic sequence Unit to shut down
- C. The APU will remain on for the duration of the flight and continue to augment the aircraft's generator electrical load in flight
- D. The remaining fuel in the left wing or fuselage tank is less than 100 pounds

2. __________ monitors the APU engine parameters and automatically controls the start sequence.
- A. The electronic sequence unit (ESU)
- B. The generator control unit (GCU)
- C. The APU fault relay panel
- D. The aircraft's electrical power monitor

3. The APU automatic shutdown feature will occur with a loss of EGT signal or a temperature that exceeds:
- A. 1,300°F
- B. 130°F
- C. 704°C
- D. Both A and C are correct

4. Prior to starting the APU, verify that _______ volts minimum are indicated on the VDC display of the aircraft's electrical power monitor.
- A. 110 VAC
- B. 28 VDC
- C. 18 VDC
- D. 28 +10 VDC

5. Operational limits of the APU include all of the following except:
- A. The APU must not be operated during aircraft fluid deicing.
- B. The APU should be started prior to landing to provide an electrical system backup should an engine generator fail.
- C. The APU must be attended while operated during pressure refueling.
- D. The APU must not be operated unattended with passengers on board.

6. An APU fire would be indicated by:
- A. The FIRE pushbutton/annunciator on the APU control panel illuminates and the aircraft master warning lights flash
- B. The Halon extinguishing agent discharges into the APU containment box
- C. The APU fire alarm in the left wheelwell sounds
- D. All of the above

7. The maximum continuous amps limitation on the APU is:
- A. 400 amps
- B. 350 amps
- C. 250 amps
- D. There is no limit

8. The APU will automatically shut down for all of the following except:
- A. Low oil pressure
- B. Loss of EGT signal
- C. An APU engine speed of <90%
- D. A GPU malfunction

CHAPTER 7
POWERPLANT

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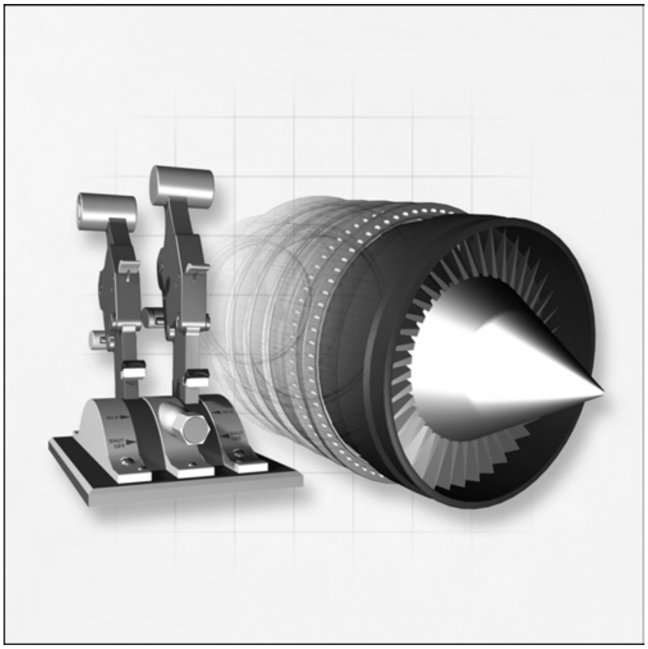
ILLUSTRATIONS

CHAPTER 7
POWERPLANT



INTRODUCTION

This chapter describes the components and operation of the Learjet 60 powerplant and related systems.

The aircraft is powered by two aft fuselage-mounted Pratt and Whitney Canada PW305A turbo-fan engines (Figure 7-1). Target-type thrust reversers, manufactured by Rohr Corporation, are a standard addition to the engines. The engines are equipped with automatic performance reserve (APR). Each engine develops 4,600 pounds of thrust at sea level at temperatures up to 29°C (84°F). With APR activated, it develops 4,600 pounds of thrust at sea level at temperatures up to 34°C (93°F).

The powerplant includes an engine fuel and control system, an ignition system, a lubricating system, accessory drive section, fire detection, fire-extinguishing, anti-ice system, and thrust reverser system.

The PW305A engine is a two-spool turbofan engine. The low-pressure spool (N_1) consists of a three-stage low-pressure turbine that drives a single-stage low-pressure compressor (fan) supported by two main bearings. The high-pressure HP spool (N_2) components are a two-stage high-pressure turbine, that drives a single-stage, high-pressure, centrifugal compressor, and a four-stage axial flow high-pressure compressor, supported by two main bearings. Variable inlet guide vanes and inlet variable stators direct the airflow to the HP compressor for efficient compressor operation. Bleed air, taken from the compressor section, is used for cabin pressurization and heating, and anti-icing. All engine-driven accessories, except for the N_1 LP rotor speed sensors, are mounted on the accessory gearbox.

Figure 7-1. Engine Installation—Learjet 60

MAJOR SECTIONS

For the purpose of explanation, the engine (Figure 7-2) is divided into eight major sections.

1. Air intake section
2. Fan section (low-pressure compressor)
3. Compressor section (high-pressure compressor)
4. Combustion section
5. Bypass duct section
6. Turbine section (LP and HP)
7. Exhaust section
8. Accessory section

AIR INTAKE SECTION

The main air intake is formed by the nacelle inlet duct. When NAC HEAT is turned on, the nacelle inlet lip is heated with engine bleed air to prevent ice accumulation.

FAN SECTION (LOW-PRESSURE COMPRESSOR)

The single-stage fan is mounted in the fan case. The case is specially constructed to reduce noise level and incorporates a Kevlar containment wrap on its exterior for the fan blade containment ring.

The fan's function is to accelerate a high airflow through fixed stators into the full-length bypass duct and to the high-pressure compressor section.

On a standard day at sea level, the airflow through the bypass duct is approximately four and one-half times that of the airflow through the compressor section. Therefore, the fan contributes the major portion of the total thrust at lower altitudes. This ratio decreases at higher altitudes as the core of the engine provides an increased portion of the total thrust. A conical (heated) spinner is attached to the front of the fan. The fan (low-pressure compressor) is driven by the three-stage LP turbine.

COMPRESSOR SECTION (HIGH-PRESSURE COMPRESSOR)

The high-pressure (HP) compressor consists of a four-stage axial flow and a single-stage centrifugal flow compressor that further increases the pressure and directs the airflow to the combustion section.

Airflow through the compressor section is also controlled by variable inlet guide vanes between the fan and the first stage of the HP compressor and by variable inlet stators between the first and second stages of the compressor. The stators are fixed between the second, third, and fourth stages of the HP compressor. The HP compressor is driven by the two-stage HP turbine.

Three compressor bleedoff valves (BOV) are located on the HP compressor case. Either channel of the electronic engine control (EEC) has the capability to control the BOVs via a dual-channel BOV solenoid valve. The solenoid valve uses HP servo air to close or open the BOVs. The bleed valves allow for efficient surge-free HP compressor operation and improves starting characteristics.

COMBUSTION SECTION

The combustion section consists of a straight-through annular flow combustion liner and outer case assembly. The combustion section includes twenty-four equally spaced fuel nozzles and two spark igniters at the four and five o'clock (aft view) positions. The combustion section controls the mixing of fuel and air, contains the combustion of gases, and directs them to the turbine section.

BYPASS DUCT SECTION

The total configuration of the inner and outer duct forms a full-length bypass duct through which bypass air is directed to the airframe bypass (exhaust nozzle) cowling.

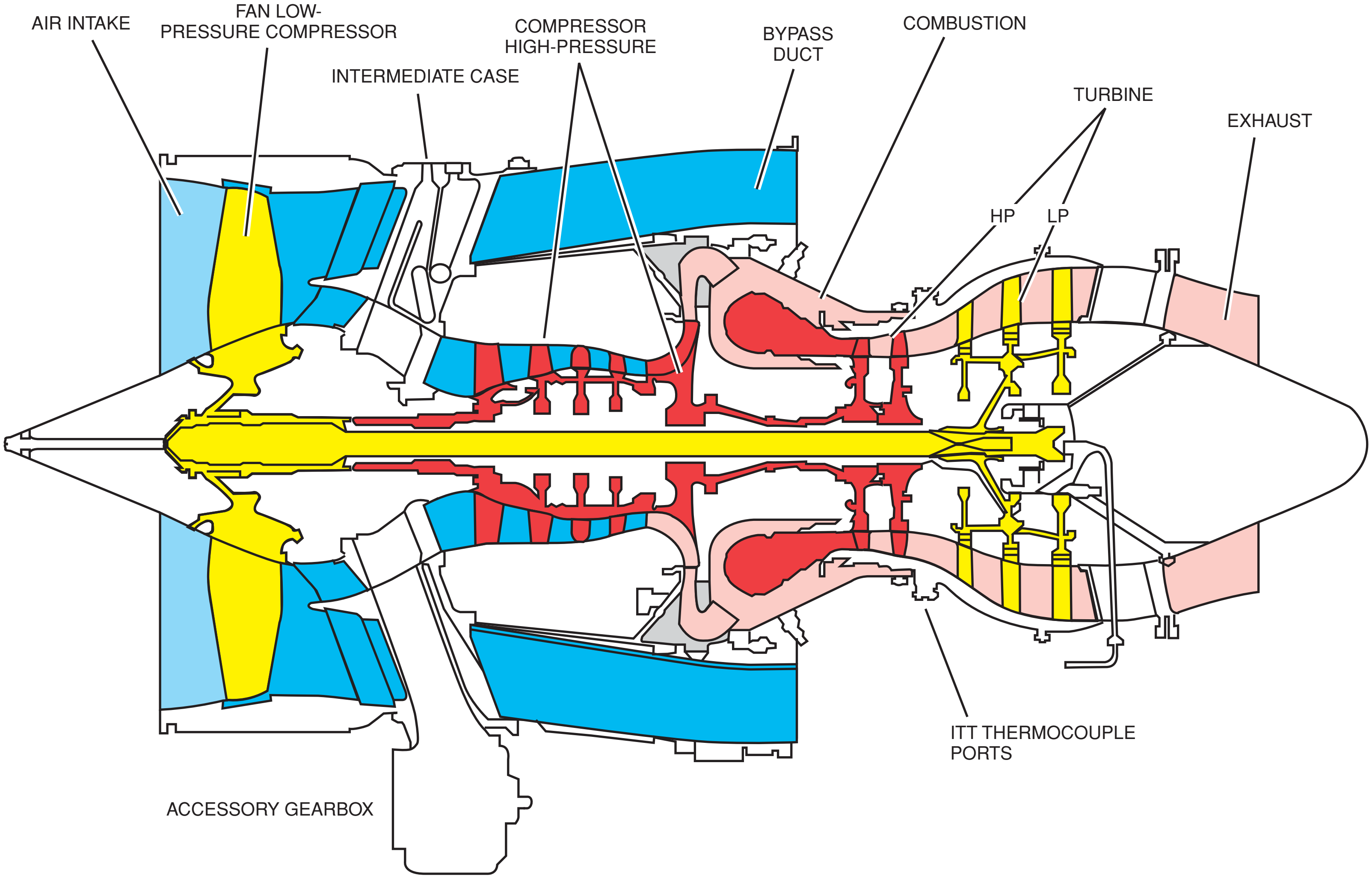


Figure 7-2. Major Engine Sections

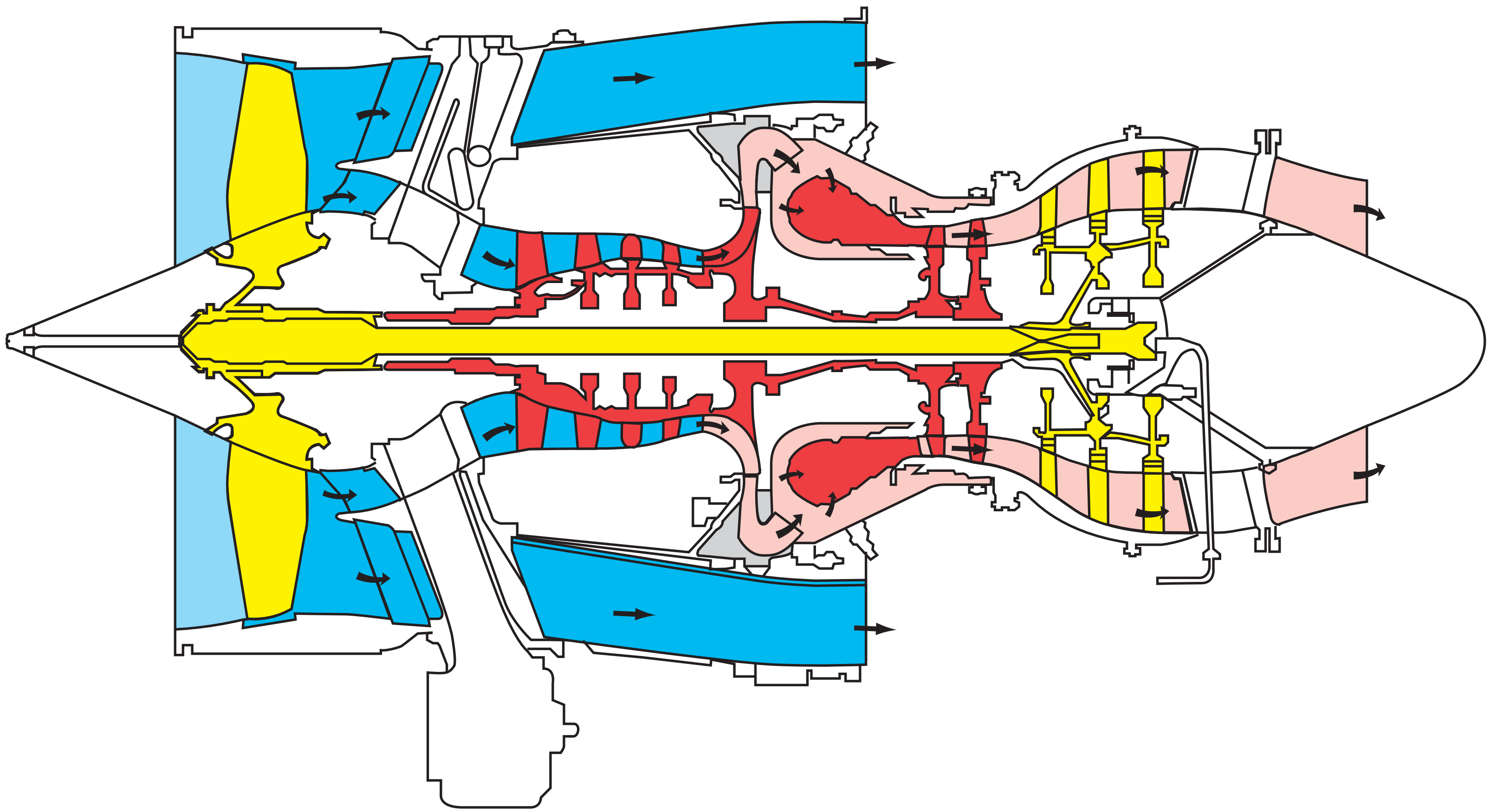


Figure 7-3. Engine Airflow Diagram

TURBINE SECTION (LP AND HP)

The two-stage, high-pressure (HP), axial flow turbines extract energy from the hot gases generated in the combustion section to drive the high-pressure compressor and the accessory gearbox (AGB). The rpm of the HP rotating group (spool) is referred to as N_2.

After the hot gases exit the HP turbine, they pass through the interturbine area where ITT temperature thermocouples are located and then into the three-stage low-pressure (LP) turbine area where additional energy is extracted. The low-pressure turbines, coupled to the low-pressure shaft, drive the low-pressure compressor (fan). The rpm of the LP rotating group (spool) is referred to as N_1 or fan.

EXHAUST SECTION

The exhaust section consists of the turbine exhaust and the bypass duct exhaust. These exhaust ducts function to direct the expanding gases to the atmosphere. These gases provide the propulsive force (thrust).

ACCESSORY SECTION

The accessory gearbox section is mounted below the engine on the intermediate case, and is driven by an angled tower shaft passing through the six o'clock strut of the intermediate case.

The following accessories are located on the accessory gearbox:

- Fuel pump
- Oil pump and scavenge segments
- Hydraulic pump
- Two N_2 probes
- Permanent magnet alternator (PMA)
- Starter/generator
- Chip Detector
- Alternator

ENGINE INSTRUMENTS

GENERAL

The engine instruments (Figure 7-4) are mounted in two vertical rows on the center instrument panel.

Figure 7-4. Engine Instruments

From top to bottom, these instruments are as follows: N_1, ITT, N_2, fuel flow, oil temperature, and oil pressure.

LOW-PRESSURE ROTOR INDICATORS

(N_1—Fan Speed)

The low-pressure rotor (fan) N_1 indicators are the primary thrust indicating instrument.

A low-pressure compressor speed (N_1) indicator for each engine is installed on the center instrument panel. Each indicator uses both a digital display and a circular scale with a pointer

to indicate low-pressure compressor (fan) N_1 or the LP rotating group rpm. The circular scale is marked from 0 to 110%, rpm in 5% increments, with an expanded scale above 80%.

The N_1 indicator also contains an N_1 rating bug. The position of the N_1 bug is computed by the EEC and transmitted to the cockpit gage. See the “Engine Control System” section of this chapter for more information on the N_1 bug.

N_1 speed is measured by two induction-type speed sensors mounted at the aft end of the LP shaft. The speed sensors are exposed to a toothed wheel mounted on the end of the LP shaft. As the teeth on the rotating LP shaft pass in front of the speed sensors, the sensors generate a frequency output signal proportional to the shaft’s rotating speed.

The two N_1 sensors on the engine have three coils in each sensor which generate a total of six separate sources of N_1 speed information. Three of the coils provide separate engine speed (N_1) information to channel A and channel B of the on-side EEC and to the N_1 indicator. Another of the coils in each sensor provide N_1 information to the overspeed circuits in each channel of the EEC. The remaining coil provides N_1 information to the opposite engine EEC for engine synchronization and APR.

Electrical power to the N_1 indicator is supplied by the left and right DC BUS 2 through the L and R N_1 circuit breakers in the ENGINE INSTR group on the left and right circuit panels. Although the N_1 indicators are powered from DC BUS 2, the circuit breakers are circled with a red ring to indicate they are powered when in the EMER BUS mode of operation. Any time electrical power is lost to the L or R DC BUS 2, the corresponding N_1 indicator(s) receive backup power from EMER BAT 1.

TURBINE TEMPERATURE INDICATORS

(ITT), $T_{4.5}$ Temperature System

The turbine temperature indicators (ITT) (Figure 7-4) are located just below the N_1 indicators on the engine instrument group. Each instrument uses both a three-place digital display and a circular scale with a pointer to indicate turbine temperature.

This circular scale is marked from 0°C to 1,000°C in 50° increments, with an expanded scale above 500°C.

The interstage turbine temperature is sensed by eight thermocouples that extend into the gas flow stream between the high- and low-pressure turbines. The thermocouples provide an average $T_{4.5}$ signal to the FADEC through a trim plug and the FADEC sends the information on to the ITT indicator.

Electrical power to the ITT indicators is 28 VDC, supplied through the L and R ITT circuit breakers in the left and right ENGINE INSTR group of circuit breakers. The L and R ITT circuit breakers receive electrical power from the L and R emergency buses respectively; therefore, ITT information is available in the EMER BUS mode of operation. With a loss of electrical power, the needle drops below 0° and the digits go blank. With a loss of data, the temperature needle still works but the digits go blank.

HIGH-PRESSURE ROTOR INDICATORS

(N_2—Turbine Speed)

The high-pressure turbine speed indicators (N_2) (Figure 7-4) are installed just below the ITT indicators in the engine instrument group.

Each N_2 indicator uses both a digital display and a circular scale with a pointer to indicate high-pressure rotor N_2 rpm. The circular scale is marked from 0 to 110% rpm in 5% increments, with an expanded scale above 80%.

N_2 speed is measured by two N_2 speed sensors that are mounted on the right side of the accessory gearbox (AGB). The sensors extend into the AGB and are exposed to the rotation of the centrifugal impeller gearshift. The AGB is driven by the N_2 high-pressure (spool) rotating group.

The two N_2 sensors each have three coils which provide a total of six separate sources of N_2 speed information. The coils in the sensors provide separate engine speed (N_2) information to CH. A and CH. B of the on-side EEC and to the corresponding N_2 indicator. They also provide N_2 speed information to the opposite engine EEC for APR and ENG SYNC circuits.

Electrical power for the N_2 indicators is 28 VDC supplied through the L and R N_2 circuit breakers in the ENGINE INSTR group. With a loss of electrical power, the needle drops below 0° and the digits go blank. The N_2 indicators do not operate in the EMER BUS mode of operation.

FUEL FLOW INDICATORS

The fuel flow indicators (Figure 7-4) are installed on the center instrument panel just below the N_2 indicators on the engine group.

Each fuel flow indicator uses both a digital display and a circular scale with a pointer to indicate fuel flow. The circular scale is marked from 0 to 3 pounds/hour x 1,000 in 50 pound/hour increments.

Fuel flow is measured by a fuel flow meter located on the left side of the engine, just below the fuel/oil heat exchanger. The fuel flow rate is transmitted to the cockpit indicator and also to the FMS(s). The fuel flow indicating system is powered by 28 VDC supplied through the L and R FUEL FLOW circuit breakers located in the ENGINE INSTR group on the left and right circuit-breaker panels. The fuel flow indicators do not operate when in the EMER BUS mode of operation.

ENGINE OIL INDICATORS

The engine oil indicators (Figure 7-4) are installed just below the fuel flow indicators.

Each engine oil indicator displays oil temperature in degrees Celsius on the the left and oil pressure in psi on the right.

Oil temperature is sensed in the oil temperature sensing port located on the left side of the engine just aft of the fuel/oil heat exchanger. Oil temperature is then transmitted to the cockpit indicator. Indicator power is from the L and R OIL TEMP-PRESS circuit breaker located in the left and right engine instrument group of circuit breakers.

Oil pressure is sensed by the main oil pressure sensor, located just above and slightly aft of the fuel filter on the right side of the engine and is transmitted to the left and right cockpit indicators. The oil pressure indicator has an expanded scale 0 to 80 psi and a compressed scale from 80 to 220 psi. The indicators are powered by the L and R OIL TEMP-PRESS circuit breakers located in the left and right ENGINE INSTR group of circuit breakers. An oil pressure switch, located by the oil pressure sensor, will cause the corresponding L or R OIL PRESS annunciator to illuminate when oil pressure is less that 20 psi. The oil pressure and temperature indicators do not operate in the EMER BUS mode of operation, but the L and R OIL PRESS annunciators will continue to operate.

ENGINE CONTROL SYSTEM

GENERAL

The engine fuel and control system pressurizes fuel routed to the engine from the aircraft fuel system and meters fuel to the combustion section of the engine. The control system also regulates the variable inlet guide vanes and variable stators and the surge bleed control system. The engine fuel system also supplies high-pressure motive-flow fuel to the aircraft fuel system for jet pump operation.

The engine control system is referred to as the FADEC (full authority digital electronic control). The acronyms “FADEC” and “EEC” are quite often used interchangeably; however, in this manual, “FADEC” is used when referring to the entire engine control system, including the thrust levers, EEC, HFCU, etc., and “EEC” is used when referring ***specifically*** to the engine computer.

The major control components of the FADEC are the thrust levers, electronic engine control (EEC), and the hydromechanical fuel control unit (HFCU). The HFCU is also sometimes referred to as the hydromechanical metering unit (HMU). In this manual it will be referred to as the HFCU.

THRUST LEVERS

Two thrust levers (Figure 7-5) are located in a quadrant on the forward section of the center pedestal. They operate in a somewhat conventional manner with full forward being maximum power; however, there is no mechanical connection between the thrust levers and the engine fuel controller as is the case with most earlier aircraft models. Each thrust lever is connected to a dual rotary variable-differential transformer (RVDT) located in the throttle quadrant. The RVDT transmits two electrical signals, one to each channel of the EEC. There is no thrust lever friction adjustment available to the pilots.

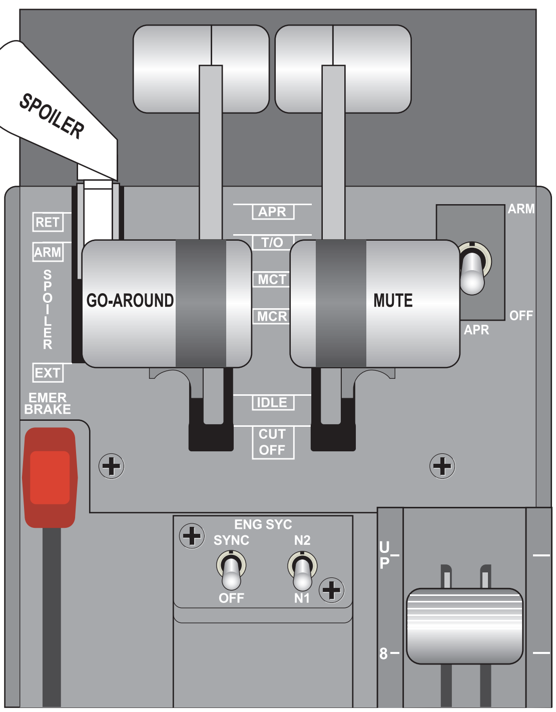


Figure 7-5. Thrust Levers

The throttle quadrant has detents and labels (Figure 7-5) at the following positions, starting full forward at APR and moving aft:

- APR (automatic performance reserve)
- T/O (takeoff thrust)
- MCT (maximum continuous thrust)
- MCR (maximum cruise)
- IDLE
- CUTOFF

The APR position is a forward limit stop rather than a detent. This position allows the crew to directly select the additional thrust that would be provided by APR, but without necessarily arming the APR system. APR is also activated with the thrust levers in the T/O detent when the APR switch is in the ARM position and the system senses an engine failure. The APR system is further described in this chapter.

When the thrust levers are positioned full forward to APR, or to one of the detents (T/O, MCT, or MCR), the FADEC computes the N_1 for that position based on ambient conditions (temperature, pressure altitude, etc.) and other EEC inputs and then signals the HFCU to meter fuel to achieve the computed N_1. Between the detents, the thrust levers are more conventional, allowing the operator to select any N_1 between idle and full power.

The IDLE position provides either flight idle or ground idle depending on the signal from the squat switches. With the thrust levers at the IDLE position, flight idle is approximately 65% N_2. Ground idle (approximately 52% N_2) will be commanded ten seconds after the aircraft touches down unless a higher engine speed is commanded through use of the thrust reversers. Stops at the idle position prevent inadvertent movement of the thrust levers to cut off. The idle stops can be released by lifting a trigger on the outboard side of each thrust lever (Figure 7-5).

When a thrust lever is placed in the CUTOFF position, a switch actuates to provide a discrete signal (overspeed) to the FADEC to initiate the engine shutdown sequence.

ELECTRONIC ENGINE CONTROL (EEC)

The EEC is on the left side of the engine toward the front (Figure 7-6). It is the electronic brain of the control system. The EEC is a dual-channel (CH. A and CH. B) computer where all engine control functions are duplicated in two independent channels.

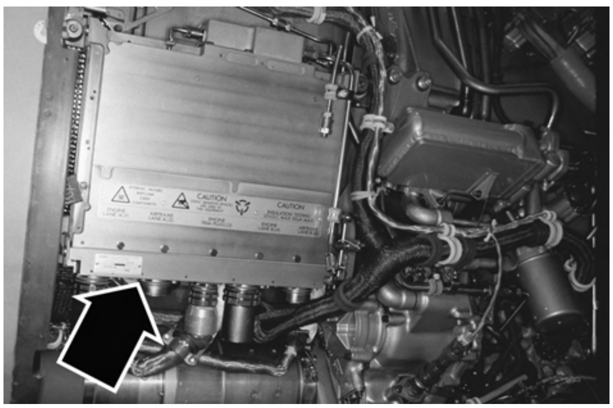


Figure 7-6. EEC On Engine

The primary EEC operating functions include:

- Thrust management
- Rating display—N_1 bug
  VIGV/IVS (VGV) control
- N_1 and N_2 overspeed protection
- Bleedoff valve control
- Starting and shutdown sequencing
- Engine speed synchronization N_1 or N_2
- APR
- $T_{4.5}$ (ITT) display

Either channel is capable of performing these functions and fully controlling the engine. During normal operation (ENG CMPTR switch in AUTO), the most capable channel is automatically selected to control the engine. The operating channel is continually compared to the standby channel and, if the operating channel becomes impaired, control of the engine will automatically transfer to the more fit of the two channels. There is no cockpit indication of which EEC channel is active.

ENG CMPTR CH.A/AUTO/CH.B SWITCHES

Although the operating channel of the FADEC is normally automatically selected, the operator also has the capability to select the channel of operation with two switches (one for each engine) on the center switch panel. These switches are labeled ENG CMPTR CH. A/AUTO/CH. B (Figure 7-7). They are lever-locking type switches with red knobs on the ends. Normally, the switches are left in the center (AUTO) position which allows the FADEC to automatically select the most capable channel. See Amber and White ENG CMPTR Light Illuminated in the Abnormal Procedures section of the *AFM* for use of these switches under abnormal conditions. Figure 7-8 shows a schematic of the engine control system.

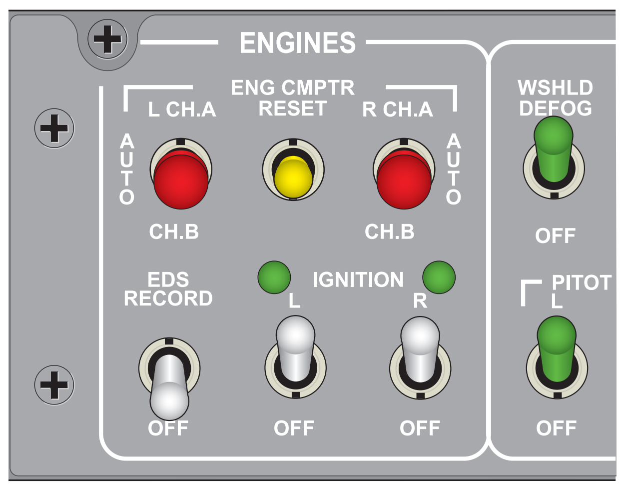


Figure 7-7. Engine Switch Panel

ENG CMPTR ANNUNCIATORS

The EEC continually monitors for internal faults and loss of necessary input and feedback signals. If an abnormal condition is detected in either channel of the EEC, it will shift control to the more healthy channel and illuminate the appropriate ENG CMPTR light(s). A white and an amber ENG CMPTR light (Figure 7-9)

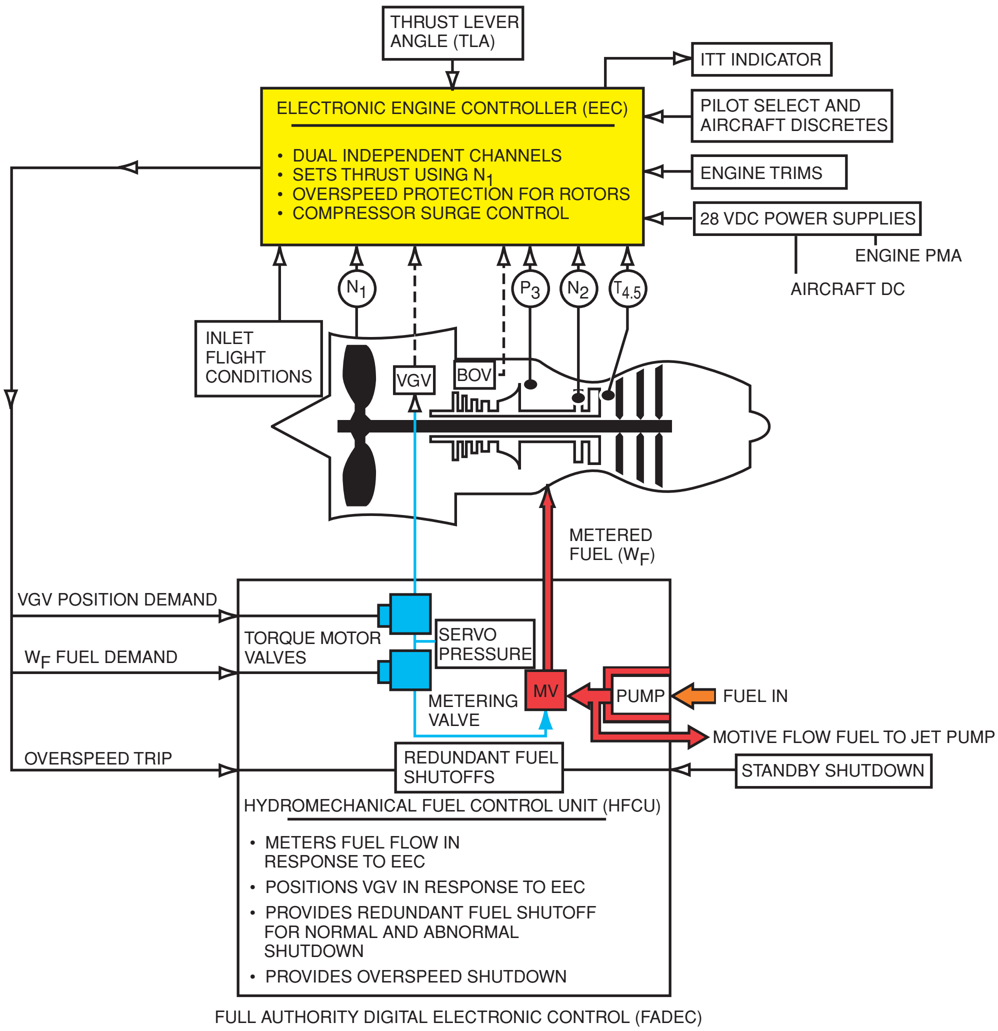


Figure 7-8. Engine Control System

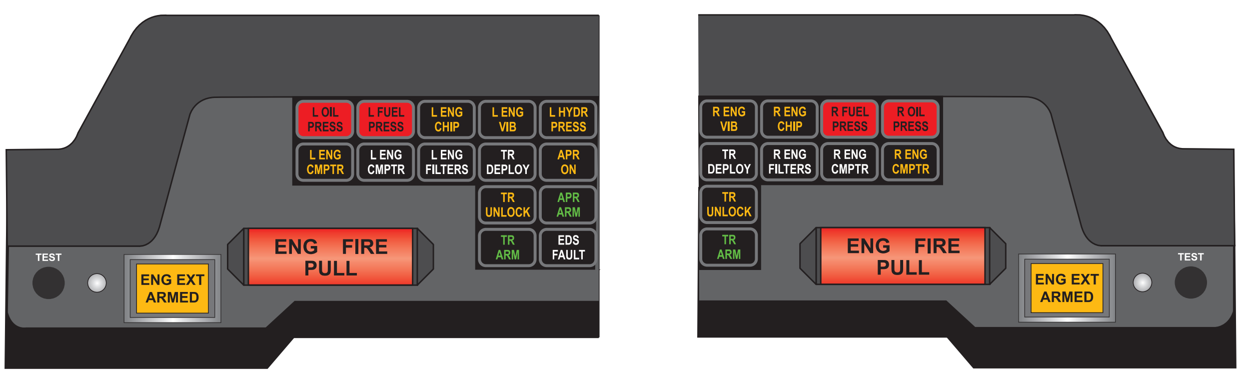


Figure 7-9. Engine Annunciators

are provided on the main annunciator panel for each engine.

- Illumination of a white ENG CMPTR light by itself indicates a minor malfunction in one or both channels of the associated FADEC.
- Illumination of an amber ENG CMPTR light by itself indicates a major malfunction in one channel of the associated FADEC.
- Illumination of both the white and amber ENG CMPTR lights may indicate major malfunction in both channels of the associated FADEC.

Dispatch is not permitted with any white or amber engine computer light illuminated. See ENG CMPTR Light Illuminated in the “Abnormal Procedures” section of the *AFM* for in-flight procedures.

ENG CMPTR RESET SWITCH

A yellow external ENG CMPTR RESET switch is located on the center switch panel between the L and R ENG CMPTR CH. A/AUTO/CH.B switches (Figure 7-7). It is spring-loaded to the center position and should be held in the left or right position (as applicable) for approximately two seconds to clear an engine computer malfunction. If the malfunction clears, the ENG CMPTR light(s) will extinguish. The reset switch is not operational on the ground and should only be used in accordance with procedures in the “Abnormal Procedures” section of the *AFM* under ENG CMPTR Light Illuminated. Aircraft 60-129 and subsequent and some prior aircraft (incorporating SB60-76-1) do not have the ENG CMPTR reset switch.

EEC INPUTS

In addition to the thrust levers, the EECs receive inputs from several other sources to provide them with the information needed to perform their functions. See Figure 7-11 for a block diagram of the following EEC inputs:

1. Pilot select and airframe discretes
2. Thrust lever angle (TLA)
3. Pneumatic signals
4. Sensors
5. N_1 and ITT trim
6. Air data computer
7. Optical link between channels

The purpose and function of most of the inputs shown in Figure 7-11 are fairly obvious. Following are some comments on the less obvious aspects of these inputs to the EEC.

The aircraft air data computers are the primary source for providing inlet static pressure (P_{AMB}) and total pressure (P_T) to the EECs. In the event of ADC failure or aircraft electrical failure, the lower P_1T_1 sensor probe (Figure 7-10) in the engine air intake duct provides this information to the EEC through a pressure transducer mounted on top of the EEC. The P_1T_1 sensor probes are the primary source for providing engine inlet total temperature (T_{TO}) to the two EEC channels. The ADCs provide a backup source of T_{TO}. Compressor discharge pressure (P_3) is also transmitted to both EEC channels.

Figure 7-10. P_1T_1 (Pressure and Temperature Probes)

When the EEC commands a change in the metering valve position in the HFCU, a linear variable differential transformer (LVDT) feeds

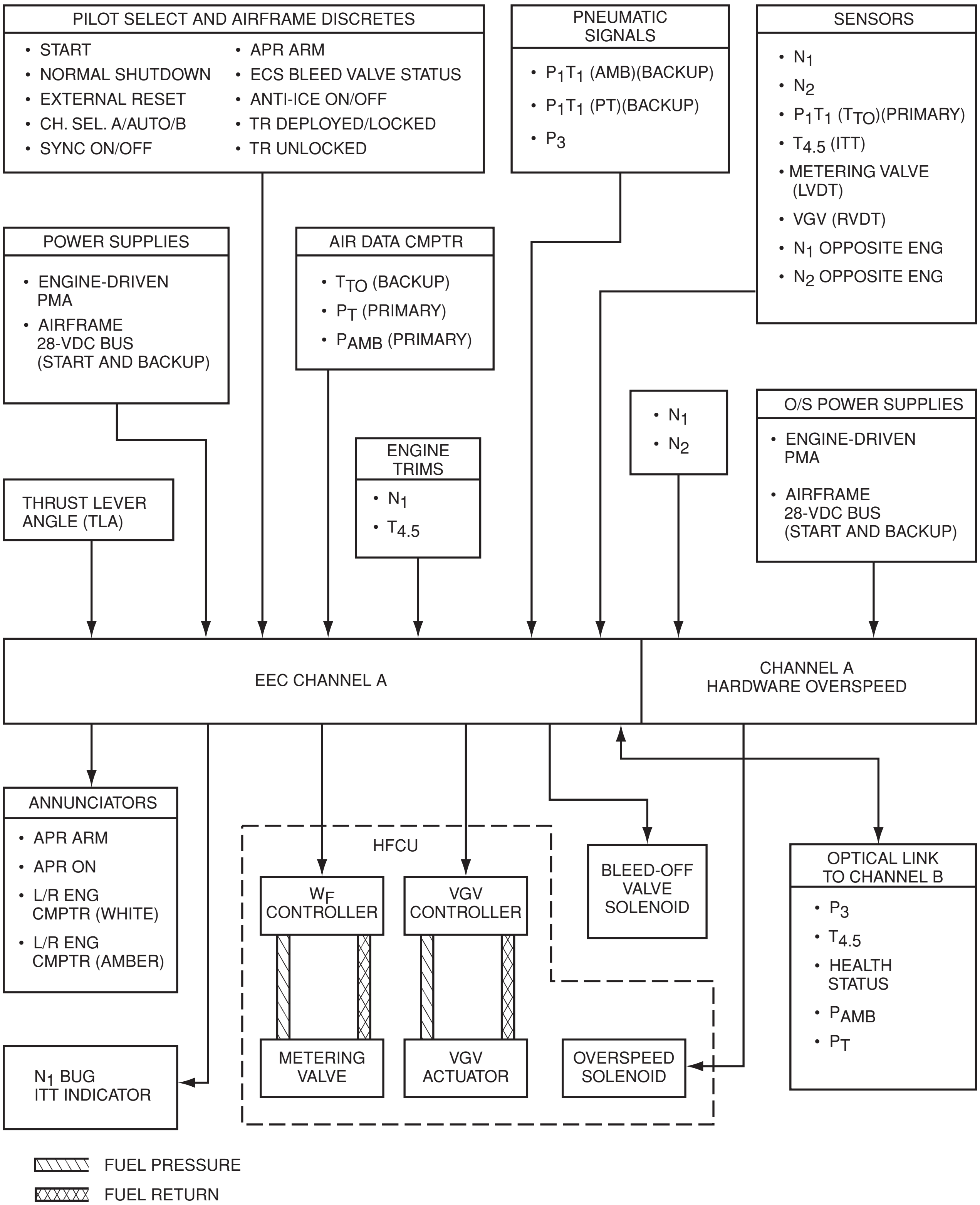


Figure 7-11. Electronic Engine Control (EEC) Schematic

back metering valve position information to each channel of the EEC. Likewise, a rotary variable differential transformer (RVDT) feeds back VGV position to each channel of the EEC.

An optical link between the two channels allows for an exchange of information between the two channels without compromising the electrical isolation between them.

ENGINE TRIMS

A nonremovable engine trim plug is tethered to the engine and connected to the EEC. This plug interacts with the EEC and biases out engine to engine variations in $T_{4.5}$ and N_1/thrust ratio to ensure that each engine provides required rated thrust (4600 lbs). The value (amount of bias) of the trim plug is determined at the factory with the engine on a test stand.

A second nonremovable trim resistor, mounted on the engine, is used to bias the cockpit N_1 gage by an amount that matches the trim plug bias used on the EEC. When N_1 engine synchronization is turned on, this trim resistor is bypassed and true, or raw, N_1 is indicated on the cockpit gage.

EEC OUTPUTS

See Figure 7-11 for a block diagram of the following EEC outputs:

1. Annunciator lights
2. $T_{4.5}$ (ITT) display and N_1 bug
3. HFCU (hydromechanical fuel control unit)
   - WF controller (torque motor valve)
   - VGV controller (torque motor valve)
   - Overspeed solenoid
4. Bleedoff valves (BOV)
5. Optical link

Annunciators that the EEC is capable of illuminating include: APR ARMED (green), APR ON (amber), and L/R ENG CMPTR (one white and one amber on each side).

The $T_{4.5}$ is input to CH. A of the EEC from eight thermocouples, located between the HP and LP turbine sections, relayed to CH. B via the optical link, and transmitted to the cockpit ITT indicator by the operating channel of the EEC.

When the FADEC has computed the N_1 based on the inputs it has received, it transmits a signal to the N_1 indicator (Figure 7-12) to set the bug to the proper position.

Figure 7-12. N_1 (Fan) Indicator

The EEC will compute N_1 bug position based on the following logic:

1. Gear or flaps down—Normal takeoff N_1, regardless of thrust lever position.
2. Gear and flaps up and thrust lever in detent—Computed N_1 for that detent.
3. Gear and flaps up and thrust lever between detents—The next higher detent rating.
4. APR is activated—Computed APR N_1.
5. Thrust reversers are deployed—Maximum reverse thrust N_1 for aircraft speed.

The output signals from the EEC to the HFCU for fuel metering, VGV control, and overspeed shutdown are further described under HFCU in this section.

The EEC electrically controls a BOV solenoid valve which uses $P_{3.0}$ servo air to pneumatically position the three BOVs located in the HP compressor case. The BOVs allow for efficient, surge-free operation of the engine throughout the speed range of the engine and improve engine starting characteristics. In the event a solenoid valve fails, the three BOVs will go to the open position.

EEC POWER SUPPLY

Aircraft DC electrical power is supplied to each channel of the FADEC through the L and R ENG CH. A and L and R ENG CH. B circuit breakers in the L and R ENGINE circuit-breaker group (Figure 7-13). The overspeed circuits in the EECs are powered separately through the L and R OVSP circuit breakers (Figure 7-13).

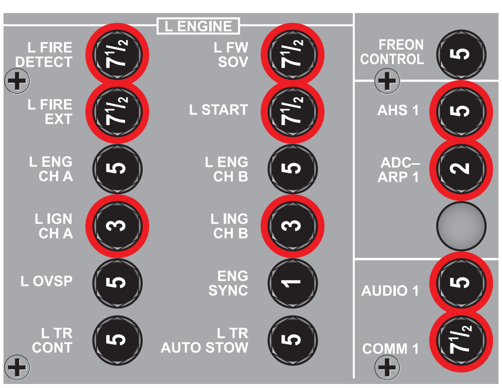
The normal aircraft DC power sources are only used for engine start and for backup in case the normal engine electrical power source fails. After the engine is started, at approximately 52% N_2, electrical power to the FADEC is normally supplied by a single, engine-driven, dual winding, permanent magnet alternator (PMA) mounted on the aft side of the accessory gearbox (AGB).

Each PMA winding supplies an EEC channel and the overspeed circuit in each channel. PMA power to the EEC does not feed through the cockpit circuit breaker. If output voltage from the PMA falls below approximately 16 VDC, the EEC automatically switches to the aircraft electrical system. There will be no indication of this switch to the pilots.

ENGINE FUEL SYSTEM

The major components in the engine fuel system (Figure 7-14) are:

- Engine fuel pump (two-stage)
- Engine fuel filter (primary)
- HFCU
  - Fuel flow (W_F) controller
  - VGV controller
  - Metering valve
  - Minimum pressure valve
  - Spill valve
  - Overspeed solenoid
  - Backup shutdown solenoid
- Fuel heater
- Flowmeter
- Dump valve
- Mechanical overspeed protection
- Fuel manifold fuel nozzles



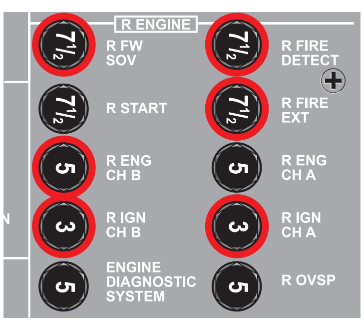


Figure 7-13. Engine Circuit Breakers

7 POWERPLANT

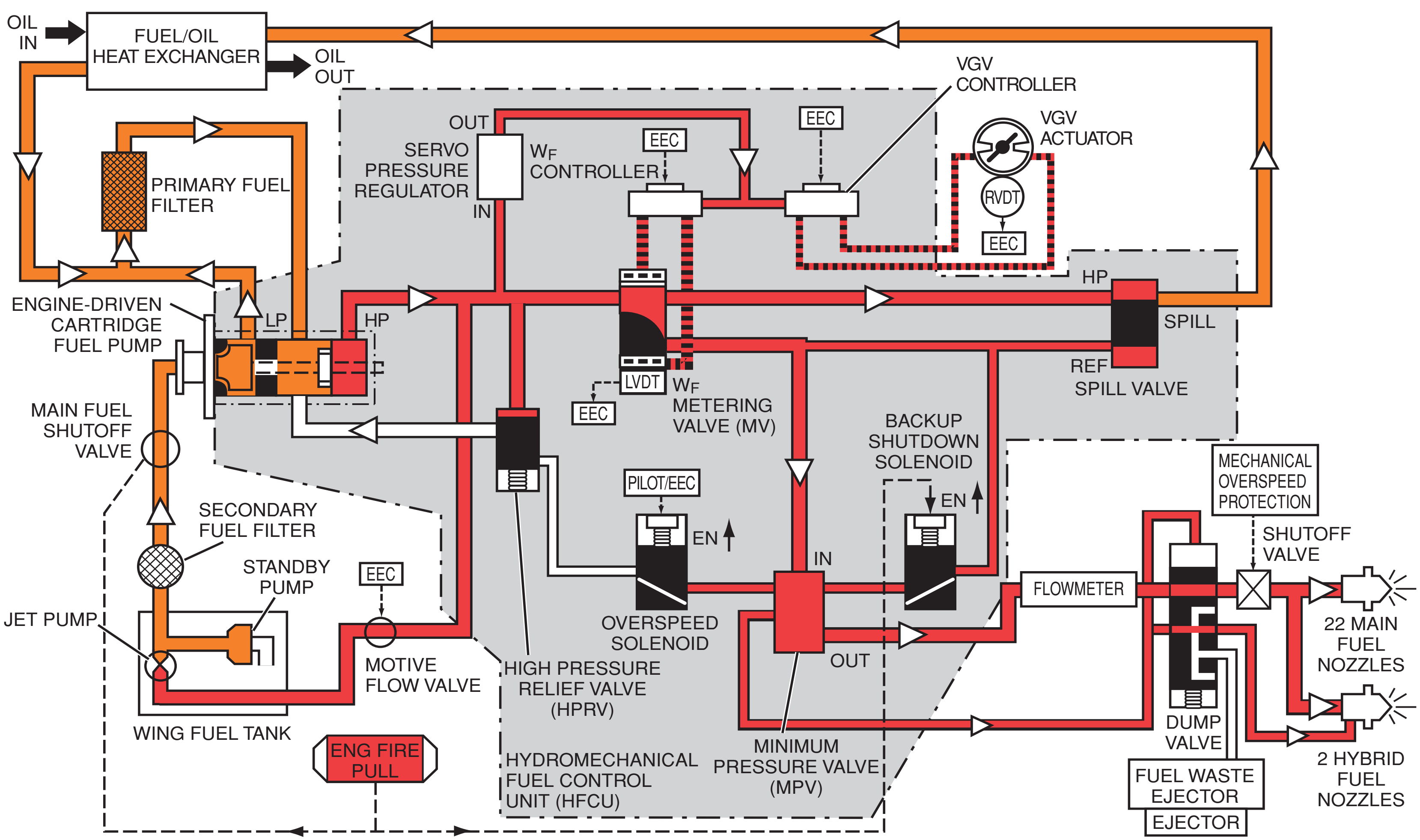


Figure 7-14. Engine Fuel System Schematic

ENGINE FUEL PUMP

The engine fuel pump is a two-stage cartridge pump that is housed within the HFCU. The mechanical drive for the pump is taken from the AGB through the PMA. The first stage of the pump is a centrifugal-type, low-pressure pump that sends fuel through the engine primary fuel filter to the inlet side of the vane-type high-pressure fuel pump. The outlet side of the second stage feeds fuel to the HFCU.

FUEL FILTER

The engine fuel filter (primary) is a cartridge-type filter that is mounted horizontally on the right side of the engine near the AGB. This filter has a bypass capability if it becomes clogged. If bypass is about to occur, an impending bypass pressure switch will cause the white L or R (as applicable) ENG FILTERS annunciator (Figure 7-9) to illuminate.

HYDROMECHANICAL FUEL CONTROL UNIT (HFCU)

The HFCU is connected to the rear face of the PMA on the aft side of the accessory gearbox (AGB). The two primary functions of the HFCU are to meter fuel flow to the fuel nozzles and to position the variable guide vanes in response to commands from the EEC. The HFCU also responds to EEC electrical inputs to the overspeed solenoid and ENG FIRE PULL T-handle inputs to the backup shutdown solenoid.

Listed below are the four electrical components in the HFCU. The first three receive inputs from the EEC and the last one is energized when the corresponding ENG FIRE PULL T-handle is pulled:

1. W_F controller
2. VGV controller
3. Overspeed solenoid
4. Backup shutdown solenoid

Fuel Flow (W_F) Controller

The fuel flow (W_F) controller uses regulated servo fuel to position the metering valve. A linear variable differential transformer (LVDT) feeds metering valve position back to both channels of the EEC. The metering valve position determines how much fuel is sent to the fuel nozzles through the minimum pressure valve.

VGV Controller

The variable guide vane (VGV) controller uses servo fuel to position the inlet variable guide vanes and inlet variable stators. A rotary variable differential transformer (RVDT) feeds guide vane position back to both channels of the EEC.

Overspeed Solenoid

The overspeed solenoid in the HFCU will be energized by the operating channel of the EEC to shutdown the engine if the EEC sees at least two indications of overspeed. One of the indications must be a hardware overspeed (110% N_1 or N_2) and the other indication may be another hardware overspeed or a software overspeed (102% N_1 or N_2). When the overspeed solenoid is energized, fuel pressure is released from the minimum pressure valve (MPV) back to the inlet side of fuel pump. This decreases the fuel pressure in the MPV causing it to shut off fuel to the fuel nozzles.

Backup Shutdown Solenoid

In the unlikely event that a malfunction of both channels of the fuel computer prevents control or shutdown of an engine, a backup method of engine shutdown is provided. If the ENG FIRE PULL T-handle is pulled, the corresponding engine main fuel shutoff valve will close and the backup shutdown solenoid in the HFCU is energized. Energizing the backup shutdown solenoid causes the spill valve to be returned which, in turn, causes fuel pressure to drop in the MPV and shuts off fuel pressure to the fuel nozzles.

Minimum Pressure Valve (MPV)

The minimum pressure valve (MPV) prevents fuel from proceeding to the fuel nozzles until the pressure reaches a certain amount. During engine start, the MPV allows fuel to flow to the two hybrid fuel nozzles when pressure reaches 335 psi, and as the start progresses, it allows fuel to the other 22 fuel nozzles when pressure reaches 445 psi.

Spill Valve and Heat Exchanger

The function of the spill valve is to maintain a constant 50 psi difference between the metering valve inlet pressure and the outlet pressure from the metering valve to the MPV. Excess fuel pressure is released by the spill valve and is routed through the fuel heater and fuel filter back to the inlet side of the HP stage of the fuel pump. The fuel heater is a simple fuel/oil heat exchanger that operates continuously when the engine is running.

FLOWMETER

The flowmeter is located between the HFCU and the fuel nozzles. It sends a signal to the fuel flow indicators in the cockpit and to the FMS(s).

FUEL DUMP VALVE AND WASTE TANK

The fuel dump valve drains fuel from the fuel manifold after engine shutdown. The fuel is collected in a fuel waste ejector and when the engine is started again, motive flow fuel from the engine fuel pump moves the residual fuel back to the inlet side of the engine fuel pump.

MECHANICAL OVERSPEED SHUTOFF

The mechanical overspeed shutoff system will shut off fuel flow to the engine should a rearward displacement of the low-pressure N_1 shaft occur.

The system is comprised of an actuating mechanism and a fuel shutoff valve. The actuating mechanism is within the exhaust case. The shutoff valve is at the six o'clock position on the outer bypass duct. The actuating mechanism is connected by a cable linkage to the fuel shutoff valve which is located between the dump valve and the fuel nozzles.

FADEC OPERATION

When the EEC receives a discrete start signal from the cockpit start switch (start/gen switch placed to START), it signals starter engagement through the GCU and closes the motive flow valve. The amber starter engage light illuminates and the JET PUMP OFF light illuminates on the fuel control panel. At 6% N_2, the EEC activates the ignition system. The EEC controls metering of fuel through the HFCU to accelerate the engine to idle rpm. The EEC controls the engine based on N_2 during start and at idle. Above idle, it computes N_1 and schedules fuel to achieve the computed N_1. As N_2 increases past 40% during start, the EEC terminates ignition. At 45% N_2, a speed switch in the starter/generator signals starter disengagement through the GCU. Internally, the starter/generator automatically switches to a generator at 45% N_2, but the generator is not connected to the electrical system until the starter generator switch is placed to the GEN position and the GPU has been disconnected.

The computer software in the EEC provides engine acceleration and deceleration schedules, thrust limiting, rpm limiting (102% N_1 and N_2), and ITT limiting. In addition, there are separate dedicated overspeed shutdown circuits in each channel of the EEC.

When CUTOFF is selected with the aircraft on the ground, the EEC first energizes the overspeed solenoid in the HFCU to shut off fuel to the fuel nozzle, and five seconds later, it signals the fuel flow (W_F) servo to close the metering valve. This sequence exercises the overspeed solenoid and would expose any dormant malfunction in the overspeed shutdown circuit or solenoid. If the thrust lever is placed to CUTOFF while airborne, the overspeed solenoid is not energized and the metering valve closes immediately.

ENGINE SUBSYSTEMS

AUTOMATIC PERFORMANCE RESERVE (APR)

The APR system provides for an automatic uptrim in thrust on the operative engine in the event of power loss on the other engine during takeoff. The system consists of a two-position (OFF–ARM) APR switch on the right side of the thrust lever quadrant (Figure 7-15), a green APR ARM and amber APR ON annunciator on the glareshield, and associated aircraft wiring. If the APR switch is placed in the ARM position prior to takeoff, the EEC checks the circuitry, and if it is satisfied with the continuity of the circuits, it will illuminate the APR ARM annunciator. If engine power is lost during takeoff, the EEC will illuminate the APR ON annunciator and increase thrust up to 7% on the operating engine but not to exceed 4,600 pounds. Loss of thrust is defined by the FADEC as meeting one or more of the following criteria:

- The N_1 differs more than 15% between engines.
- The N_2 differs more than 7.5 % between engines.
- The N_1 differs more than 4% between engines, and the N_1 on the low power engine is decreasing at a rate greater than 5% per second.
- The N_2 differs more than 2% between engines, and the N_2 on the low power engine is decreasing at a rate greater than 2% per second.

Should automatic activation of APR fail to occur, or if it is not armed, APR thrust can be manually obtained by setting the thrust lever(s) to the APR position. In this case, the APR ON light will not illuminate. Once invoked, either manually or automatically, the APR thrust schedule will remain active until the APR switch is set to OFF.

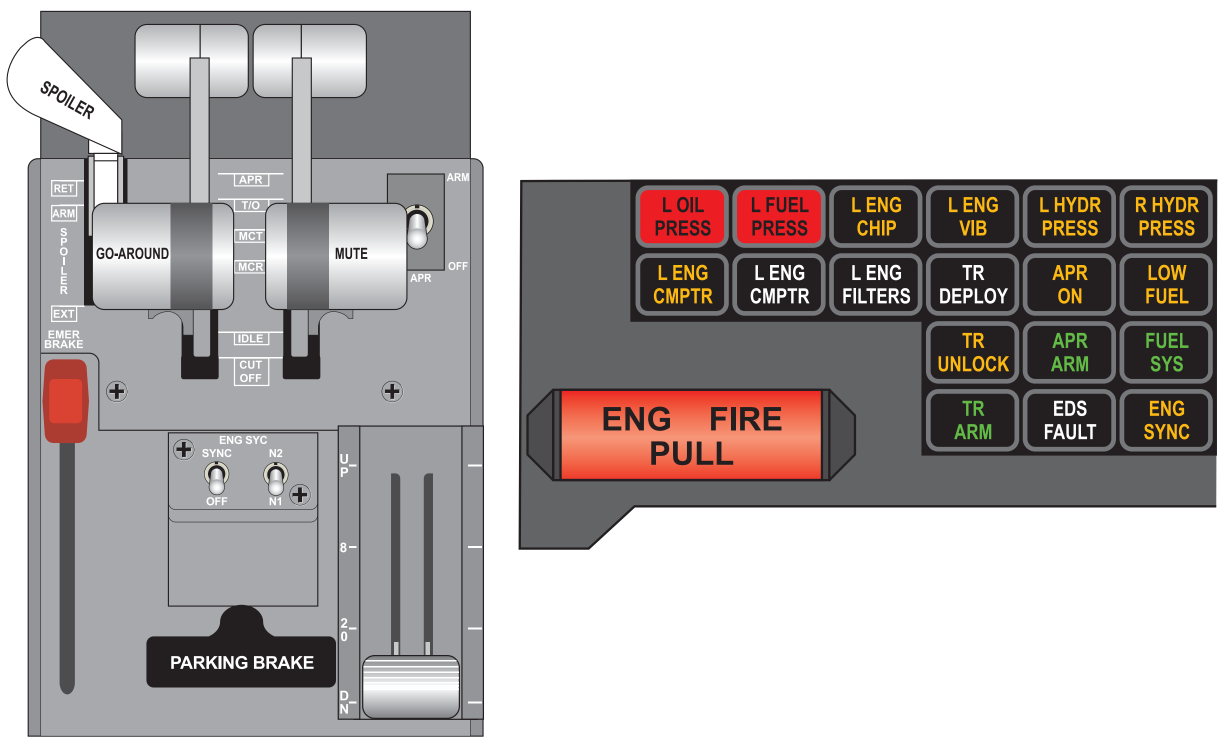


Figure 7-15. APR and ENG SYNC Switches and Annunciators

ENGINE SYNCHRONIZER

Engine synchronization can be selected in flight with a two-position OFF-SYNC switch and a N_1–N_2 switch located on the thrust lever quadrant (Figure 7-15). Also included in the system is an amber ENG SYNC annunciator, on the glareshield, and engine synchronization circuits within the FADECs. Electrical power for the system is through the ENGINE SYNC circuit breaker on the left circuit-breaker panel.

Synchronization is accomplished by maintaining the N_1 or N_2 (as selected) speed of the slave engine in sync with the master engine. The master engine is determined and so designated during installation (Figure 7-16). The following criteria must be satisfied before the system will operate:

- ENG SYNC switch set to SYNC.
- The difference between N_1 or N_2 (whichever is selected) is no more than 5%.
- Thrust levers are in the range from IDLE to MCT.
- Thrust reversers are stowed.
- APR is disarmed.
- Squat switch is in the air mode.

When N_1 is selected for synchronization, the trim to the N_1 indicator is removed; therefore, the N_1 and N_1 bug presentation will reflect actual N_1 speed.

The engine sync system should not be used during takeoff or landing or during single engine operation. If the SYNC switch is on and the nose gear is not up and locked, the amber ENG SYNC light on the glareshield will be illuminated.

ENGINE VIB LIGHTS

Illumination of either L ENG VIB or R ENG VIB light (Figure 7-9) indicates an abnormally high level of vibration in the associated engine. The lights are activated by a signal conditioning box located in the tailcone. A transducer installed on the top of each engine's intermediate case provides the trigger to illuminate the corresponding L ENG VIB or R ENG VIB light. The system is powered through the L and R ENG VIB MON circuit breakers in the left and right engine group of circuit breakers.

IGNITION SYSTEM

The PW 305A engine incorporates a high-energy ignition system. Each engine ignition system consists of an air ignition switch, a green

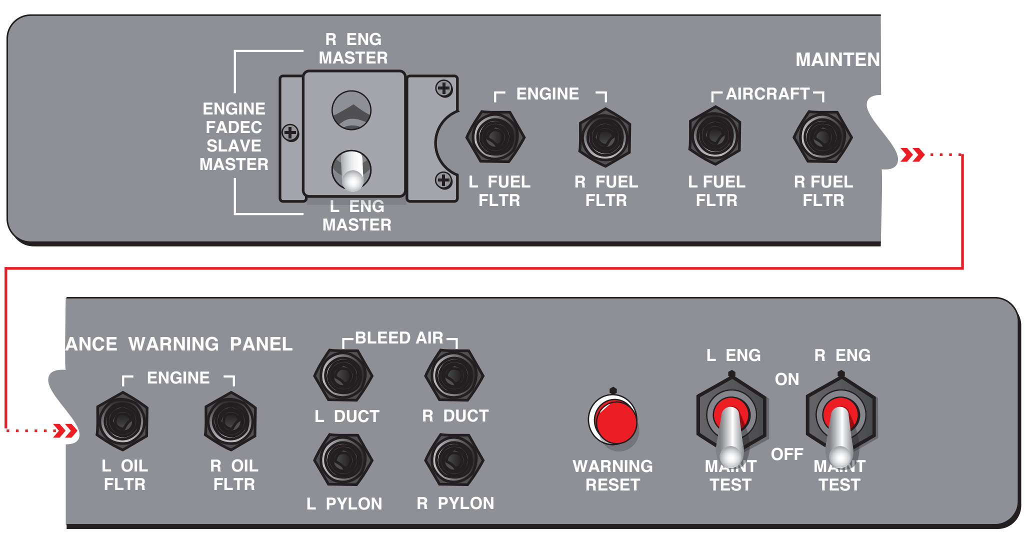


Figure 7-16. Maintenance Warning Panel (Tailcone)

annunciator light, a FADEC control circuit, two ignition exciter units, and two spark igniters.

When an ignition system is activated, the dual ignition exciter units receive electrical power and the spark igniter plugs fire simultaneously. During a normal starter-assisted start cycle, the ignition system is automatically energized by the FADEC at approximately 6% N_2 (green ignition light illuminates) and automatically deenergized at approximately 40% N_2 (green ignition light extinguishes).

Green annunciator lights (Figure 7-17) above each ignition switch indicate ignition operation. The corresponding light will be illuminated whenever the associated dual ignition units are activated. Illumination of these lights, however, does not necessarily guarantee that the spark igniter plugs are firing. The ignition lights incorporate a dimming circuit which causes the lights to dim when the NAV LTS switch is set to ON.

At pressure altitudes above 20,000 feet, the FADEC will automatically command ignition if the thrust lever is placed to cutoff and then returned to the idle position. If the engine restarts, the FADEC will then terminate ignition at 40% N_2.

At pressure altitudes below 20,000 feet and with the thrust lever at IDLE or above, the FADEC will automatically energize the ignition system if N_2 drops below 40%.

The ignition systems can be manually activated with the ignition switches (Figure 7-17) for continuous ignition during takeoff and landing or when flying through heavy rain, icing, or turbulent air conditions. The systems are capable of continuous operation and do not have a duty cycle limit.

Electrical power for ignition is 28VDC supplied through the L and R IGN CH. A or L and R IGN CH. B circuit breakers in the left and right engine group of circuit breakers (Figure 7-13).

STARTERS

A starter/generator is installed on the accessory gearbox (AGB) of the engine. The starter for each engine is powered from the battery charging bus through a single-start contactor (relay) for each starter (Figure 7-18). Placing the START/GEN switch to START sends a discrete signal to the FADEC, and the FADEC initiates the start sequence if the corresponding thrust lever is in the IDLE position. The FADEC signals the starter relay to close through the corresponding generator control unit (GCU). When engine rpm reaches 45% N_2, a speed sensor in the starter generator signals the GCU to disengage the starter relay. The starter relay also disengages if the START/GEN switch is place to OFF prior to 45% N_2. The amber start light is illuminated whenever power is supplied to the associated starter.

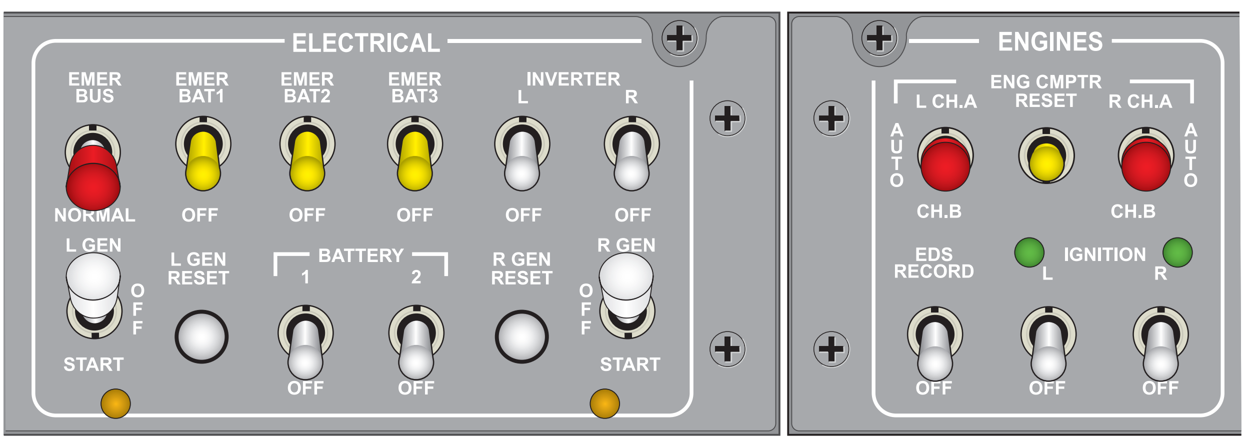


Figure 7-17. Start/Ignition Switches and Lights

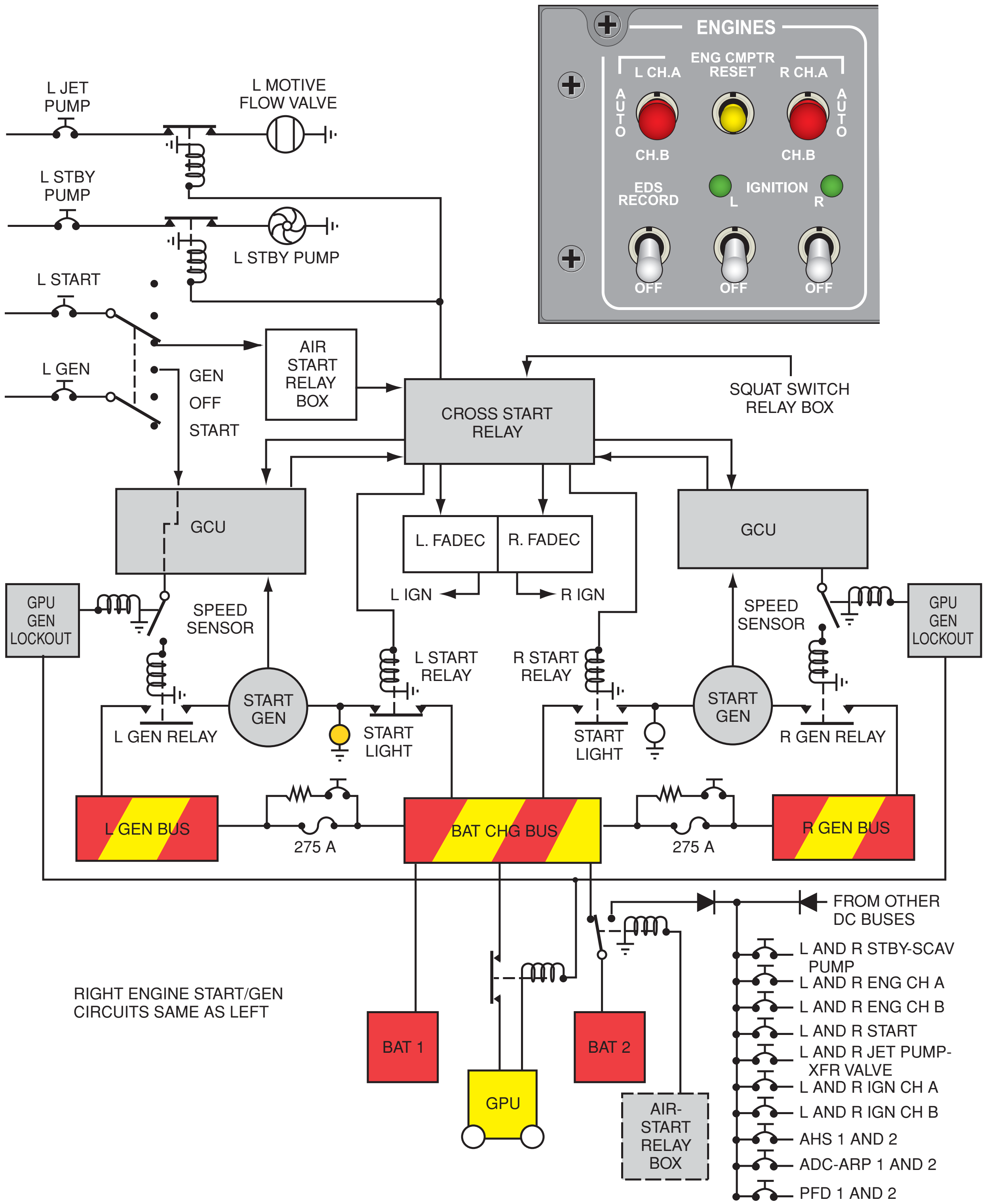


Figure 7-18. Engine Starter/Generator Schematic

The GEN–OFF–START switches, on the electrical panel (Figure 7-17), are the lever-lock type. They must be pulled out to move to the START position. They do not have to be pulled out to move to the OFF or GEN position.

In addition to the starter, a number of other circuits are energized when the start switch is placed to START.

The standby fuel pump in the associated wing is energized, the FADEC closes the motive flow valve and initiates the start sequence, and the following are depowered if turned on: air conditioner, auxiliary heat, and stabilizer heat.

An air-start relay box allows certain equipment to receive electrical power directly from aircraft BAT 2 during a starter-assisted airstart. This prevents voltage fluctuations to critical or voltage-sensitive equipment (Figure 7-18).

A cross start relay allows an engine to be started using current from the opposite engine generator through the operating engine's generator relay and starter relay. This reduces the electrical load on the generator relay and the 275-amp current limiter. The cross start relay will only function with the aircraft on the ground.

ENGINE OIL SYSTEM

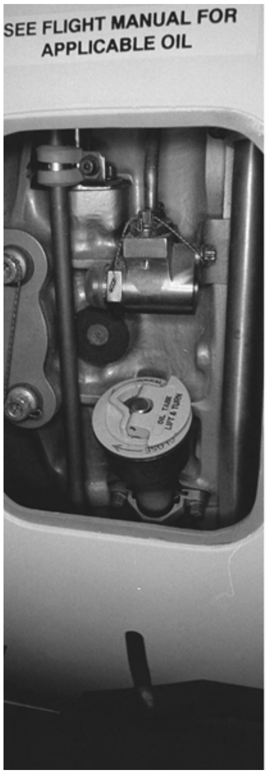
General

The engine oil system provides for cooling, lubrication, cleaning, and anti-icing to various engine components, bearings, and accessory gearbox.

The oil system is a pressure-scavenge system consisting of a single oil pressure pump, three scavenge pumps, oil tank, oil filter, and fuel/oil heat exchanger. Oil is filtered and cooled enroute to the lubricating points in the engine.

Oil Supply

The oil supply is contained in a tank that is integral to the intermediate engine case, capacity 1.95 U.S. gallons. The filler neck and cap are located beneath an access panel on the outboard side of each engine. The filler cap incorporates a dipstick to check the oil level (Figure 7-19). The oil level should be checked during the aircraft exterior inspection. See the Oil Addendum in the *AFM* for proper servicing and approved oils.



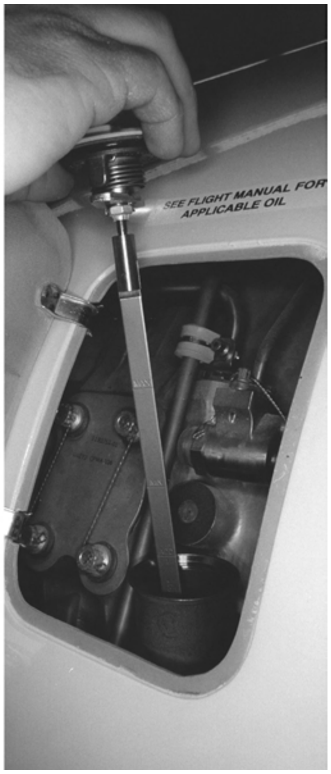


Figure 7-19. Oil Filler Neck and Cap

System Operation

Oil is drawn from the oil tank (Figure 7-20) by the pressure pump and is routed through pressure adjusting, filtering, and temperature control components before delivery to the engine for lubrication. It passes through the oil filter which has a bypass capability if the filter should become clogged. If bypass is impending due to a clogged filter, the L or R (as applicable) ENG FILTERS light will illuminate on the glareshield annunciator panel. After passing through the filter, oil is routed through the fuel/oil heat exchanger. This component serves two purposes: it cools the engine oil and heats the fuel going to the HP stage of the engine fuel pump.

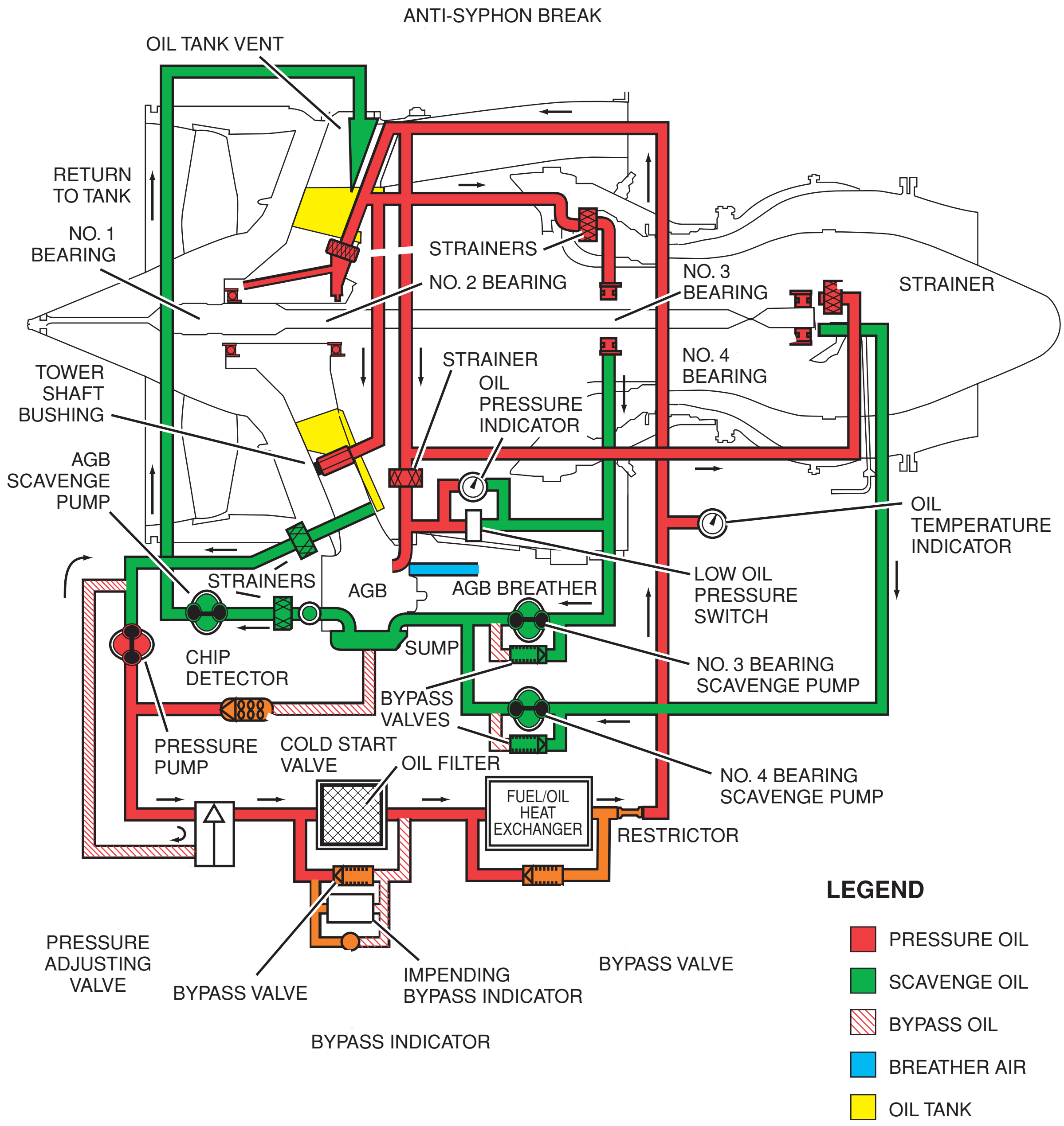


Figure 7-20. Engine Oil System Schematic

The pressure oil is now delivered to the four main engine bearings, the tower shaft, and the accessory gearbox. Scavenging is accomplished by two pumps that return the oil to the accessory gearbox and a third scavenge pump that pumps the oil from the AGB back to the reservoir.

A magnetic chip detector is installed in the AGB scavenge pump line enroute to the reservoir for detection of metal particles in the oil. Cockpit indication is via the amber L or R ENG CHIP light (Figure 7-21) on the glareshield annunciator panel.

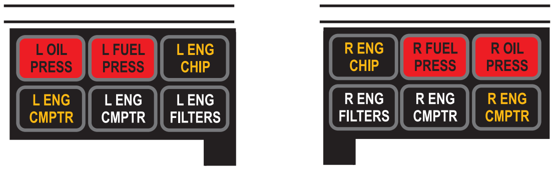


Figure 7-21. OIL PRESS and ENG CHIP Annunciators

All bearing, gearbox, and oil tank areas are vented by the scavenge lines back to the oil reservoir where it is vented overboard through the AGB driven centrifugal air/oil separator.

Oil Temp/Press Indication

Engine oil pressure is monitored by a transducer that drives the cockpit indicator (Figure 7-22) and by a pressure switch connected to the red L or R OIL PRESS light (Figure 7-21) on the annunciator panel.

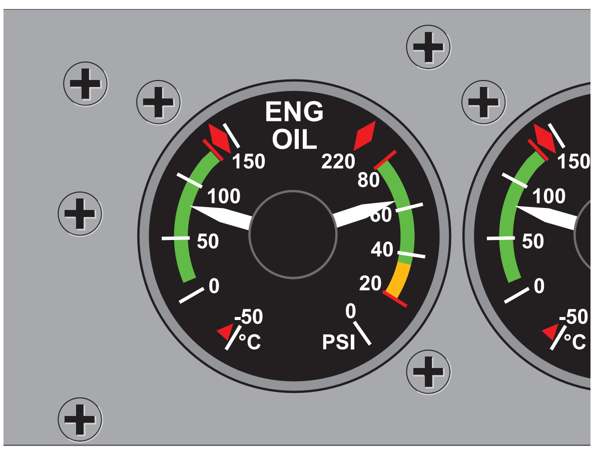


Figure 7-22. Oil Temperature and Oil Pressure Indicators

Engine oil temperature is sensed in the pressure oil line enroute to the bearings and AGB. See "Engine Instruments" in this chapter for more on the oil temperature and pressure indicators.

See Section I of the *AFM* for operating limits on the oil temperature and pressure.

ENGINE DIAGNOSTIC SYSTEM

An engine diagnostic system (EDS) is installed to provide engine fault recording and trend monitoring. The system continually records engine parameters and allows the crew to record conditions at anytime through use of the EDS RECORD switch on the pilot's switch panel. Normal use of the system entails downloading data from the EDS and submitting to Pratt and Whitney Canada for analysis on a monthly basis. The EDS is intended for maintenance functions only and not for in-flight monitoring or diagnosis by the flight crew.

A white EDS FAULT light, located on the glareshield annunciator panel (Figure 7-15), is included in the engine diagnostics system. Illumination of the light indicates one of the following:

- The EDS has lost power.
- The EDS has a built-in test equipment (BITE) failure.
- The EDS memory is 85% full.
- The system has detected an engine condition which is out of acceptable parameters.

The diagnostic system includes an EDS RECORD switch located on the center switch panel. The purpose of the switch is to allow the flight crew to initiate data collection by the EDS. When the spring-loaded switch is momentarily actuated, the engine parameters existing four minutes prior to and one minute after switch actuation will be recorded in the EDU memory. The unit's capacity allows approximately 200 hours of data storage.

THRUST REVERSERS

GENERAL

Thrust reversers, manufactured by the Rohr Corporation are installed on all model 60 Learjets. They use hydraulically actuated target-type doors to redirect the engine exhaust gases to produce reverse thrust. The thrust reversers are controlled with two thrust reverser levers mounted on top of the thrust levers and operation is annunciated by a green TR ARM, amber TR UNLOCK, and white TR DEPLOY for each engine. There is no thrust reverser control panel.

The thrust reversers are electrically controlled through the thrust reverser levers, the FADECs, and two (one for each engine) thrust reverser relay boxes (Figure 7-23) located in the tailcone. A throttle balk solenoid is located in the thrust lever quadrant. It prevents movement of the thrust reverser levers aft of idle reverse (approximately 40% N_1) unless both engine thrust reversers are deployed.

The thrust reversers are hydraulically actuated and rely on aircraft main hydraulic system pressure for operation. There are two hydraulic control units (HCUs) in the tailcone (one for each engine). They each contain a mechanical shutoff valve, an electrically controlled isolation valve, and an electrically actuated control valve.

The thrust reverser hydraulic actuators and an upper and lower target-type blocker door are attached to the engine's aft section. The doors are faired with the nacelle and form the engine afterbody. Two hydraulic actuators, one on each side of the engine, move a four-bar linkage attached to each door to reposition them. The linkage has an overcenter feature (referred to as primary latch) that will prevent the blocker doors from deploying in event of hydraulic fluid loss. Also, aircraft 60-041 and subsequent and previous aircraft modified by SB 60-78-2 are equipped with two secondary latches which hold the upper and lower blocker doors closed. The secondary latches are hydraulically released when thrust reverser deployment is commanded.

There are two unlock switches and a deploy switch attached to the aft end of the thrust reverser support beam. These switches control thrust reverser sequencing and cause the associated amber TR UNLOCK and white TR DEPLOY annunciators to illuminate. The secondary latches also contain switches that will cause the UNLOCK light to illuminate in the event of an uncommanded unlatching of an associated thrust reverse blocker door.

SYSTEM OPERATION (NORMAL)

In order for thrust reversers to deploy, the following prerequisite conditions must exist:

- TR CONT and TR AUTO STOW circuit breakers must be in
- Aircraft on the ground (both squat switches in ground mode)
- Applicable thrust lever in IDLE position
- Hydraulic pressure available from either engine-driven hydraulic pump

When the above conditions are satisfied, the green TR ARM light will illuminate, indicating the relay box has opened the isolation valve in the HCU and that pressure is available up to the control valve in the HCU.

Lifting the thrust reverser lever to the deploy detent will initiate deployment. The relay box will position the control valve in the HCU to send hydraulic pressure to the deploy side of the hydraulic actuators and to the secondary latches (if installed) to release them. As soon as the main hydraulic actuators begin to move from stow toward deploy, the stow switches will trip causing the TR UNLOCK annunciator to illuminate and signal the engine FADEC to limit thrust to idle power.

NOTE

Thrust reverser arming may be delayed during soft landings until both squat switches are in the ground mode.

Excessive force applied to the thrust reverser levers may prevent the balk solenoid from releasing.

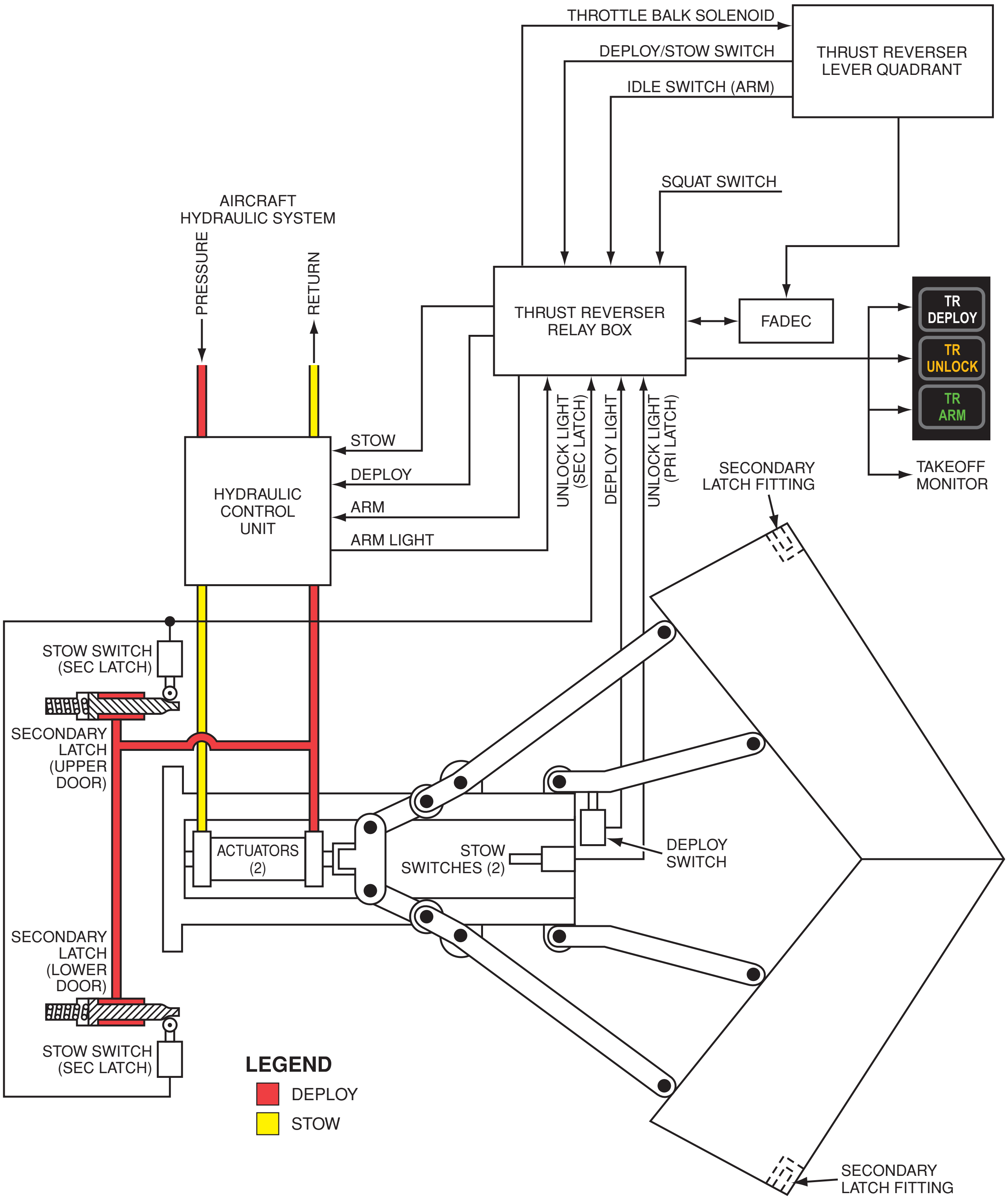


Figure 7-23. Thrust Reverser System Schematic

When the blocker doors reach the fully deployed position, the deploy switch is tripped by one of the idler arms causing the white TR DEPLOY annunciator to illuminate and the TR UNLOCK light to extinguish. When the TR UNLOCK light extinguishes, a signal to the engine FADEC will allow engine thrust to increase above idle. The N_1 bug will reposition indicating the FADEC is utilizing the reverse schedule. When both TR DEPLOY lights are illuminated, the throttle balk solenoid is released allowing the operator to move the TR levers aft to accelerate the engines. The FADEC will limit reverse thrust as a function of airspeed (provided by ADC 1 and 2), decreasing thrust as the aircraft slows down. See Figure 7-24 for schedule of maximum reverser thrust. If ADC airspeed is not provided to the FADEC, the maximum reverse thrust available will be 65% N_1. After using reverse thrust, normal stow is accomplished by returning the thrust reverser levers to the STOW position. While the doors transition from deploy to stow, the white TR DEPLOY light will extinguish and the amber TR UNLOCK light will illuminate. When the doors reach the fully stowed and locked position (both primary and secondary [if applicable] latches engaged), the amber TR UNLOCK light will extinguish. See the "Limitations" section of the *AFM* for thrust reverser operating limitations.

SYSTEM OPERATION (ABNORMAL)

The thrust reverser system includes an automatic stow feature which minimizes the possibility of an inadvertent deployment. If the system senses an unlocked primary latch in flight, the automatic stowing sequence is initiated. Also, if a TR UNLOCK light illuminates, the FADEC will reduce engine thrust to idle unless the TR UNLOCK light was caused by a secondary latch switch.

If a TR UNLOCK and/or TR DEPLOY annunciator illuminates during takeoff while the aircraft is still on the ground, the FADEC will reduce engine thrust to idle and the takeoff warning horn will sound. If the unlocked condition is caused by a secondary latch the FADEC will not reduce engine thrust to idle and the takeoff warning horn will not sound.

If a TR UNLOCK and/or TR DEPLOY annunciator illuminates in flight, the FADEC will reduce engine thrust to idle and autostow will be commanded if the UNLOCK indication was from the primary latch. If the indication was due to a secondary latch, the FADEC will not reduce thrust or command autostow. The green TR

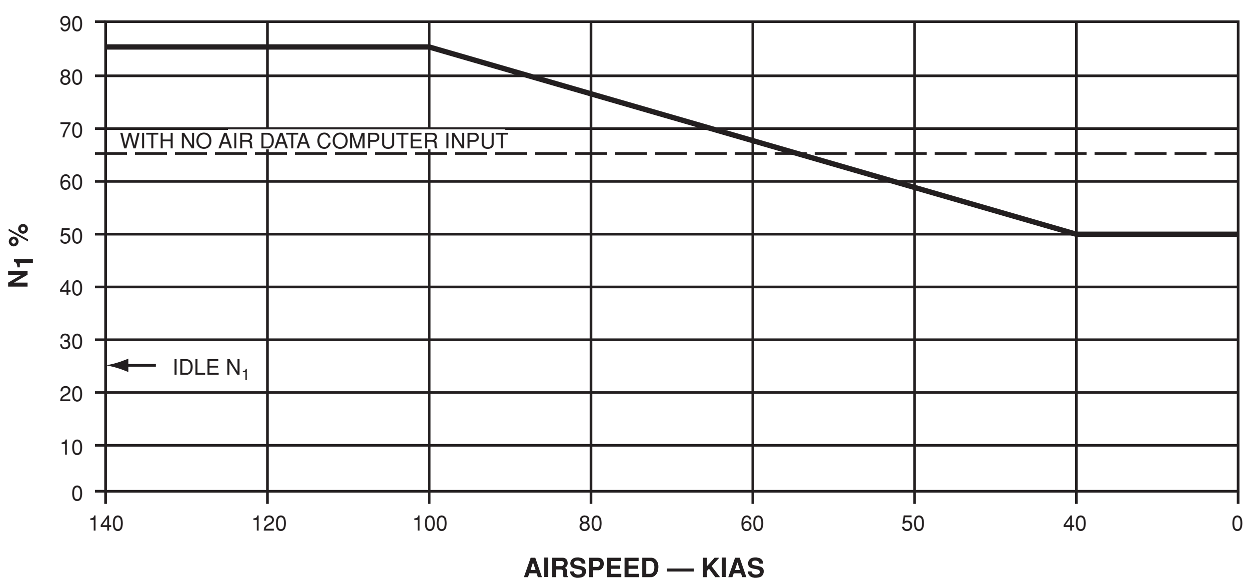


Figure 7-24. Reverse Thrust Power Schedule

ARM light will flash while autostow is applying hydraulic pressure to stow the thrust reverser and both the white and amber ENG CMPTR lights will illuminate.

The thrust reverser on one engine can be deployed on a single landing; however, the throttle balk solenoid will not allow anything other than idle reverse unless both DEPLOY lights illuminate. A thrust reverser can be deployed with the engine shutdown if the prerequisite conditions listed exist. If making a single engine landing, both thrust reversers can be deployed and full reverse can be used on the operating engine. Use of rudder/nose steering/differential braking would be required to maintain directional control.

See Abnormal and Emergency Procedures sections of the *AFM* for more information on thrust reverser abnormal/emergency procedures.

NOTES

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QUESTIONS

1. The PW 305A engines each provide 4,600 pounds of thrust at:
A. Sea level up to 29.0°C normal
B. Sea level up to 34.0°C maximum
C. Sea level at any temperature
D. Both A and B are correct.

2. The engine LP spool (N_1) consists of:
A. A four-stage axial flow compressor driven by a two-stage turbine
B. A single-stage fan and a single-stage centrifugal compressor driven by a three-stage turbine
C. A single-stage low-pressure compressor (fan), driven by a three-stage turbine
D. A single-stage fan and a four-stage axial flow compressor driven by a two-stage turbine

3. The starter/generator, in the start mode through the accessory gearbox, is applied to the:
A. LP spool (rotor)
B. HP spool (rotor)
C. Fan (low-pressure compressor)
D. LP turbine

4. The engine HP spool (rotor) N_2 consists of a:
A. Three-stage axial compressor and a four-stage radial turbine
B. Single-stage centrifugal compressor and a two-stage axial turbine
C. Two-stage axial compressor and a single-stage centrifugal compressor driven by a two-stage axial flow turbine
D. Four-stage axial flow compressor and a single-stage centrifugal compressor driven by a two-stage axial flow turbine

5. The engine instruments N_1 and N_2 are:
A. Self-generating tachometers
B. AC-powered through the 26-VAC buses
C. DC-powered, (hot wired to battery), with battery switches OFF
D. DC-powered through CBs on the DC Bus 2/EMER BAT and DC BUS 1 respectively

6. The engine oil temp/pressure indication requires:
A. 28 DC Power
B. 115 VAC
C. 26 VAC
D. Self-generating

7. The primary engine thrust indicating instrument is the:
A. High speed rotor N_2
B. ITT
C. Low speed rotor N_1
D. Fuel flow

8. The maximum ITT during engine start is:
A. 850°C
B. 870°C for 2 seconds
C. 795°C for 20 seconds
D. 950°C

9. The maximum transient ITT during takeoff is:
A. 800°C for 25 seconds
B. 825°C for 20 seconds
C. 900°C
D. 850°C for 2 seconds

10. On the ground illumination of the L or R ENG FILTERS light indicates:

A. Impending bypass of aircraft fuel filter
B. Impending bypass of engine oil filter
C. Impending bypass of engine fuel filter
D. Any of the above

11. During engine operation the BOVs, (bleed-off valves) are positioned by:

A. EEC command channel
B. Remains closed
C. Remains at 1/3 open position
D. Has no function

12. During FADEC operation of the engine, engine overspeed protection is provided by:

A. N_1 and N_2 overspeed signals for the active EEC channel to the overspeed solenoid in the HFCU
B. Mechanical overspeed protection system
C. Engine overspeed protection is not required in a FADEC controlled engine.
D. Both A and B are correct.

13. Which of the following statements regarding fuel control is true in the event of aircraft total electrical failure?

A. Engine shuts down because aircraft DC power is not available to the EEC.
B. All engine control functions remain operational as electrical power to the EEC is provided by a PMA in the accessory gearbox.
C. Backup aircraft DC power to the EEC is not available.
D. Both B and C are correct.

14. The ENG SYNC light indicates that:

A. ENG SYNC is not turned ON, or it has failed.
B. ENG SYNC is operating properly.
C. ENG SYNC is turned ON, and the nose landing gear is not in the up and locked position.
D. The engines are synchronized.

15. A left or right ENG CMPTR white light indicates:

A. A minor malfunction has occurred in one or both channels of the corresponding EEC.
B. EEC will continue to monitor for the healthiest channel to control the engine.
C. On ground—do not dispatch. In flight—refer to the “White ENG CMPTR Light Illuminated” procedure in Section IV of the *AFM*.
D. All of the above

16. A left or right ENG CMPTR amber light indicates:

A. A major malfunction has occurred in the one channel of the EEC.
B. FADEC will select the healthiest channel to control the engine.
C. On ground—do not dispatch. In flight—refer to the “Amber ENG CMPTR Light Illuminated” procedure in Section IV of the *AFM*.
D. All of the above

17. During a normal ground start, the ignition light should come on when:

A. The thrust lever is moved to idle.
B. The START–GEN switch is moved to start.
C. N_2 reaches 6%.
D. N_1 reaches 10%.

18. Engine oil pressure and oil temperature indications require ___________ electrical power.

A. 26 VAC
B. 115 VAC
C. Both DC and AC
D. DC

19. The ENG CMPTR switches are normally in the auto position. Selection of the CH. A or CH. B position in flight is permitted when:

A. "Amber ENG CMPTR Light Illuminated" procedure is used, Section IV, *AFM*.
B. "Amber and White ENG CMPTR Light Illuminated" procedure is used, Section IV, *AFM*.
C. "White ENG CMPTR Light Illuminated" procedure is used, Section IV, *AFM*.
D. Selection of CH. A or CH. B position is not allowed in flight—ground use only.

20. The green TR ARM light means:

A. The TR CONT and TR AUTO STOW CBs are in.
B. The aircraft is on the ground (squat switches in GND MODE).
C. The engine thrust lever is in idle.
D. All of the above conditions exist.

21. As the thrust levers are advanced for takeoff, the TR ARM lights:

A. Go out
B. Stay ON
C. Flash
D. Both TR ARM and TR DEPLOY lights flash.

22. During single-engine operation which of the following statements is/are true in regard to using both thrust reversers?

A. Both engine thrust levers must be in idle.
B. The full range of reverse thrust will be available on the operating engine with both T/Rs deployed.
C. Neither thrust reverser can be deployed.
D. Both A and B are correct.

23. If a thrust reverser fault occurs in flight and a reverser unlocks, it is stowed by:

A. The autostowing feature through the TR AUTO-STOW CB
B. Selecting emergency stow
C. There is no AUTO STOW on this reverser.
D. The thrust reverser warning horn sounds.

24. Maximum reverse thrust is usable down to:

A. 60 KIAS
B. 50 KIAS
C. 25 KIAS
D. 10 KIAS

CHAPTER 8
FIRE PROTECTION

CONTENTS

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CHAPTER 8
FIRE PROTECTION

INTRODUCTION

The Learjet 60 engine fire protection system is composed of a fire detector sensing element and two fire detection control units located in the tailcone, two engine fire indicator lights in the cockpit (one for each engine), two fire-extinguisher bottles that can be activated from the cockpit, and a fire detector and extinguisher circuit test.

ENGINE FIRE DETECTION AND INDICATORS

Each engine cowling contains three heat sensing elements connected to two fire detection control units in the tailcone and two red ENG FIRE PULL lights (one for each engine) on the glareshield annunciator. The fire detection control unit (Figure 8-1) monitors the electrical resistance of three sections of sensing elements. Two of the sections, one around the pylon firewall and another under the accessory gearbox, require a temperature of approximately 300°F to activate the fire detection control unit. The third section is around the rear nacelle area and requires a temperature of approximately 700°F to activate the fire detection control unit. When the sensing element is heated to the above temperatures, the

electrical resistance in the sensor is reduced, allowing sufficient current to flow between the conductor and the tube activating the fire detection control unit which causes the red ENG FIRE PULL T-handle to flash.

Electrical power for the system is 28 VDC, supplied through the L and R FIRE DETECT circuit breakers located in the left and right engine circuit-breakers group. L & R FIRE DETECT is also operative in EMER BUS mode of operation.

ENG FIRE PULL LIGHTS

The red ENG FIRE PULL warning lights are part of the T-handles, one located at each end of the glareshield annunciator panel (Figure 8-2). In the event of an engine fire, the warning light in the T-handle will flash until the fire or overheat condition no longer exists. Operation of the T-handle is explained under Engine Fire Extinguisher.

FIRE DETECTION SYSTEM TEST

The rotary SYSTEM TEST switch (Figure 8-3) on the center instrument panel is used to test the fire detection system. Rotating the switch to FIRE DET and depressing the TEST PRESS button initiates three separate tests. These are:

1. Both ENG FIRE PULL T-handles should flash, validating the continuity of the engine heat sensing elements (four lights).
2. Both ENG EXT ARMED lights should illuminate, validating the continuity of the fire extinguisher circuits (two lights).
3. Both the L and R BLEED-AIR annunciators should illuminate, validating the continuity of the overheat detection sensor in the tailcone (two lights) (see Chapter 9, Pneumatics for more information on the BLEED-AIR annunciators).

Both master warning lights will also illuminate. A valid test should show all ten lights illuminated.

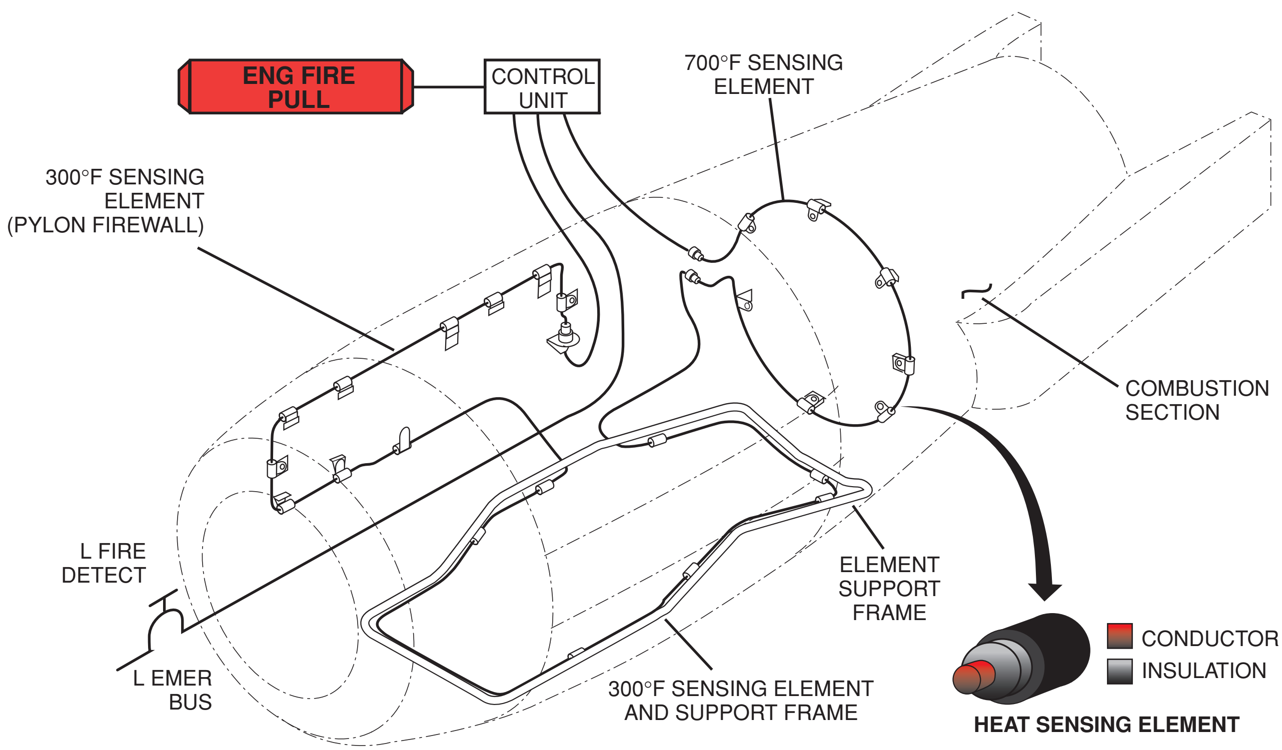

Figure 8-1. Engine Fire Detection System

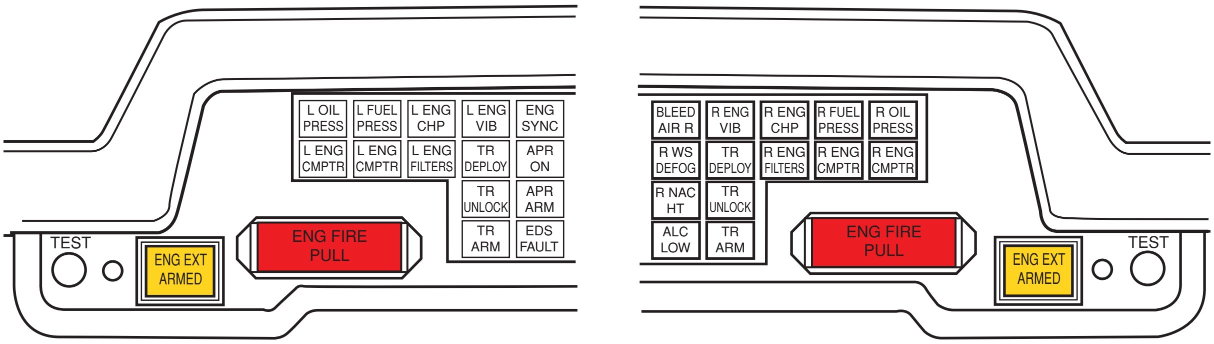

Figure 8-2. Engine Fire Indicator Lights and Controls

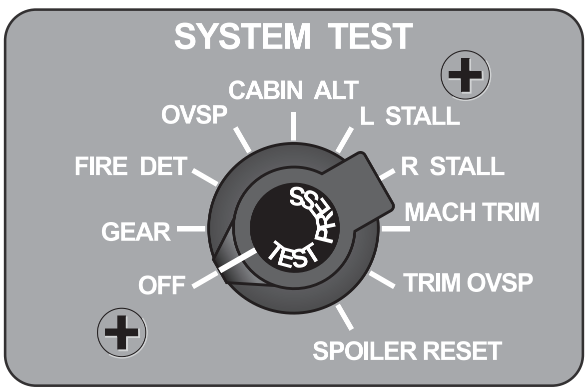

Figure 8-3. System Test Switch

ENGINE FIRE EXTINGUISHER

EXTINGUISHER CONTAINERS

The spherical, extinguishing agent containers are located in the tailcone area. The containers are plumbed to each engine cowling and provide the airplane with a two-shot system. The containers are charged with Halon 1301 (CF2Br) bromotrifluoromethane. Halon 1301 is nontoxic at normal temperatures and is noncorrosive. Therefore, no special cleaning of the engine or cowling area is required if the system has been used. The agent is stored under pressure, and a pressure gage is installed on each container. The pressure gages indicate approximately 600 psi at 70°F when the containers are properly serviced (Figure 8-4).

A thermal relief valve on each container is plumbed to a common thermal discharge port on the outside of the fuselage below the left engine pylon. The thermal relief valves will release bottle pressure at approximately 220°F.

ENG FIRE PULL T-HANDLES AND ENG EXT ARMED LIGHTS

When an ENG FIRE PULL light begins to flash, it indicates a fire or overheat condition in the associated engine. Following *Airplane Flight Manual* procedures, the pilot first places the affected engine thrust lever to idle. If the fire continues more than 15 seconds or there are other indications of a fire, the pilot places the affected engine thrust lever to cutoff and then pulls the corresponding T-handle. Pulling the T-handle closes main fuel, hydraulic, and bleed-air shutoff valves for that engine, sends a shutdown signal to the backup shutdown solenoid in the HFCU, and arms the fire extinguishers (Figure 8-4). Electrical power to close these valves and arm the extinguishers is provided through 28 VDC, L and R FW SOV and L and R FIRE EXT circuit breakers in the left and right engine circuit-breaker group. The system remains operative in the EMER BUS mode.

Fire-extinguisher arming is indicated by illumination of the ENG EXT ARMED lights adjacent to the ENG FIRE PULL T-handles (Figure 8-2). Depressing an illuminated ENG EXT ARMED light momentarily supplies 28 VDC to an explosive cartridge that discharges and allows the contents of one fire-extinguisher container to flow into the associated engine nacelle.

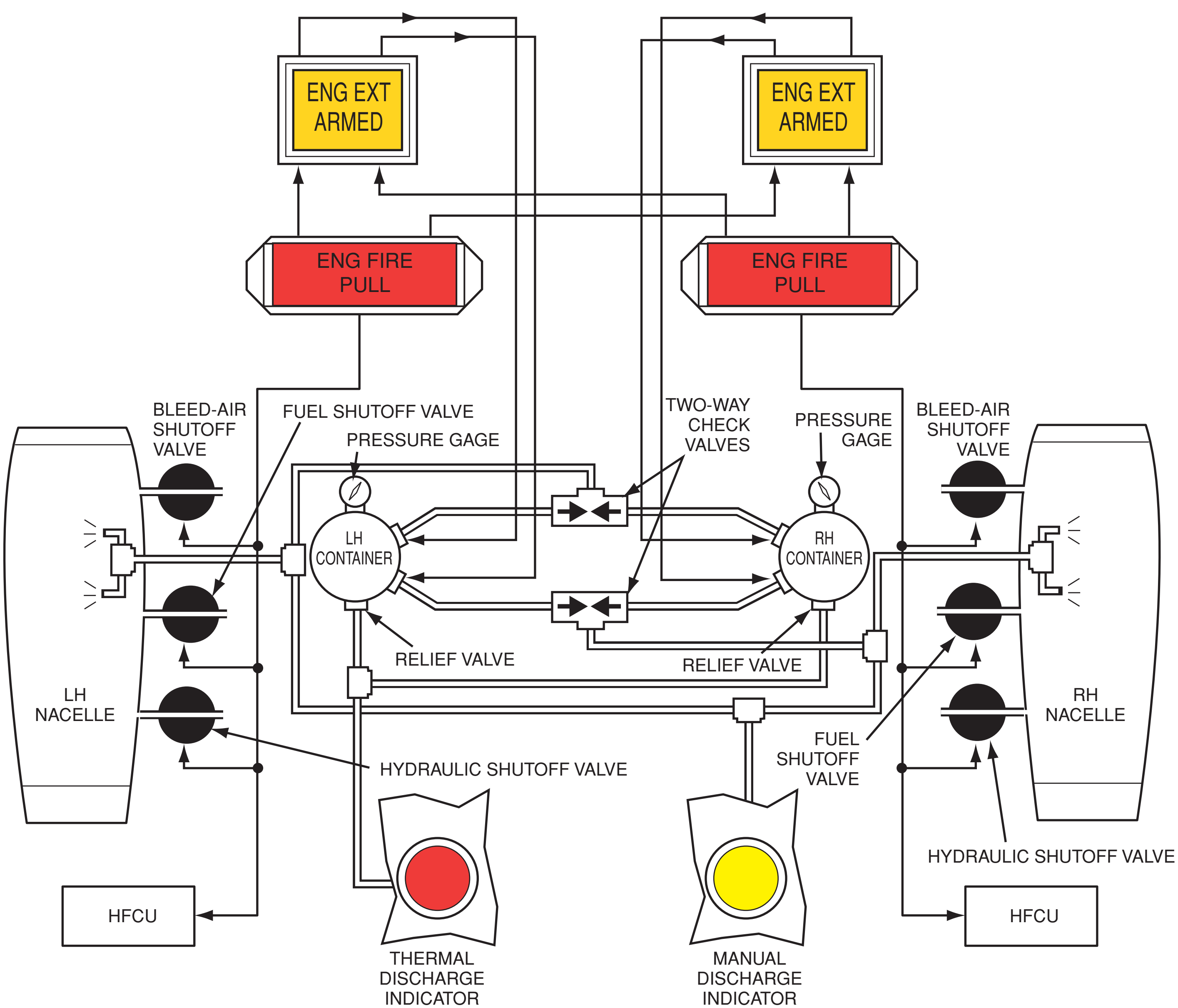

Figure 8-4. Engine Fire-Extinguisher System

When the ENG EXT ARMED light is depressed, a holding relay also is engaged, which turns out the ENG EXT ARMED light, indicating the associated bottle has been discharged. Either ENG EXT ARMED light may be depressed to extinguish the fire. Should one bottle control the fire, the other container is still available to either engine.

EXTINGUISHER DISCHARGE INDICATORS

Two colored disc-type indicators are flush-mounted in the side of the fuselage, below the left engine pylon (Figure 8-5). The forward, red-colored disc covers the thermal discharge port. It will be ruptured if one or both pressure relief valves has released bottle contents overboard. The aft, yellow-colored disc will rupture if either bottle is discharged by depressing the illuminated ENG EXT ARMED light. The integrity of the two discs is checked during the external preflight inspection.

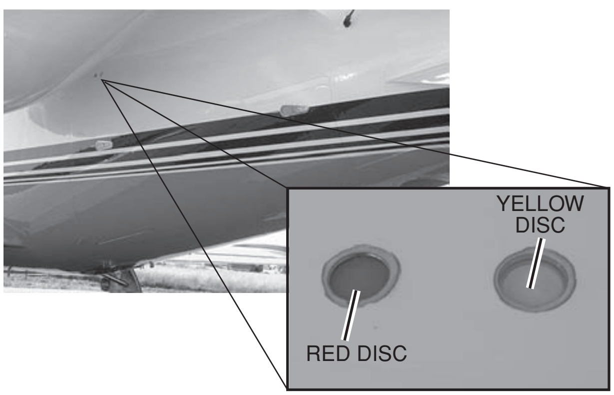

Figure 8-5. Fire-Extinguisher Discharge Indicators

CABIN SMOKE DETECTION

A smoke detector is located in the aft cabin baggage area (Figure 8-6).

Electrical power for operation is supplied through the CABIN FIRE DETECT circuit breaker on the copilot's CABIN group of circuit breakers. When activated, an amplified signal is sent to illuminate and flash the red CABIN FIRE warning light.

Figure 8-6. Portable Fire Extinguisher and Smoke Detector

HAND-HELD FIRE EXTINGUISHERS

Two hand-held fire extinguishers provide for interior fire protection. Both extinguishers are located in the cockpit, one on each side. They are stored in brackets, mounted behind the oxygen mask container, on each side (Figure 8-6). Some aircraft may have these fire extinguishers stored in different locations.

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QUESTIONS

1. Engine fire-extinguisher containers are located in:

A. The nacelles
B. The engine pylons
C. The tailcone
D. The baggage compartment

2. The power-off preflight of the engine fire extinguishers includes:

A. Checking the condition of one yellow and one red blowout disc
B. Checking the condition of two yellow and two red blowout discs
C. Checking blowout discs and extinguisher charge gages (all on the left side of the fuselage)
D. Activating the SYSTEM TEST switch to FIRE DET

3. When the left ENG FIRE PULL T-handle is pulled:

A. It discharges one extinguisher into the left nacelle.
B. It closes the main fuel, hydraulic, and bleed-air shutoff valves for the left engine, signals the HFCU, and arms both fire extinguishers.
C. It discharges one extinguisher and arms the second.
D. It ruptures the yellow discharge indicator disc.

4. There is a smoke detector:

A. Located in the tailcone baggage compartment
B. Located in each of the cabin and tailcone baggage compartments, all tied to the same CABIN FIRE warning light
C. In the cockpit and in the cabin baggage compartment
D. In the cabin baggage area, which illuminates the red flashing CABIN FIRE annunciator

CHAPTER 9
PNEUMATICS

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CHAPTER 9
PNEUMATICS

INTRODUCTION

The aircraft pneumatic systems consist of two independent engine bleed-air distribution systems. The bleed-air pneumatic systems provide engine bleed air for anti-icing, air conditioning, and pressurization.

GENERAL

The bleed-air distribution systems on each engine are connected by ducting in the tailcone to three (2HP and 1LP) manifolds (Figure 9-1). One HP manifold distributes high-pressure (HP) bleed air to the engine anti-ice system, the alcohol anti-ice system, bleed-air temperature control system, and the pressurization control system. The other HP manifold provides HP air on an as needed basis to supplement the wing anti-ice system (refer to Chapter 10—"Ice and Rain Protection"; Chapter 11—"Air Conditioning"; and Chapter 12—"Pressurization"). The remaining manifold distributes a mixture of low-pressure (LP) and HP bleed air to the cabin pressurization system, the air-conditioning system, the anti-icing systems, and the hydraulic reservoir. The distribution systems are referred to as high-pressure (HP) and low-pressure (LP) systems, even though the LP system does, at times, use some HP bleed air (Figure 9-1).

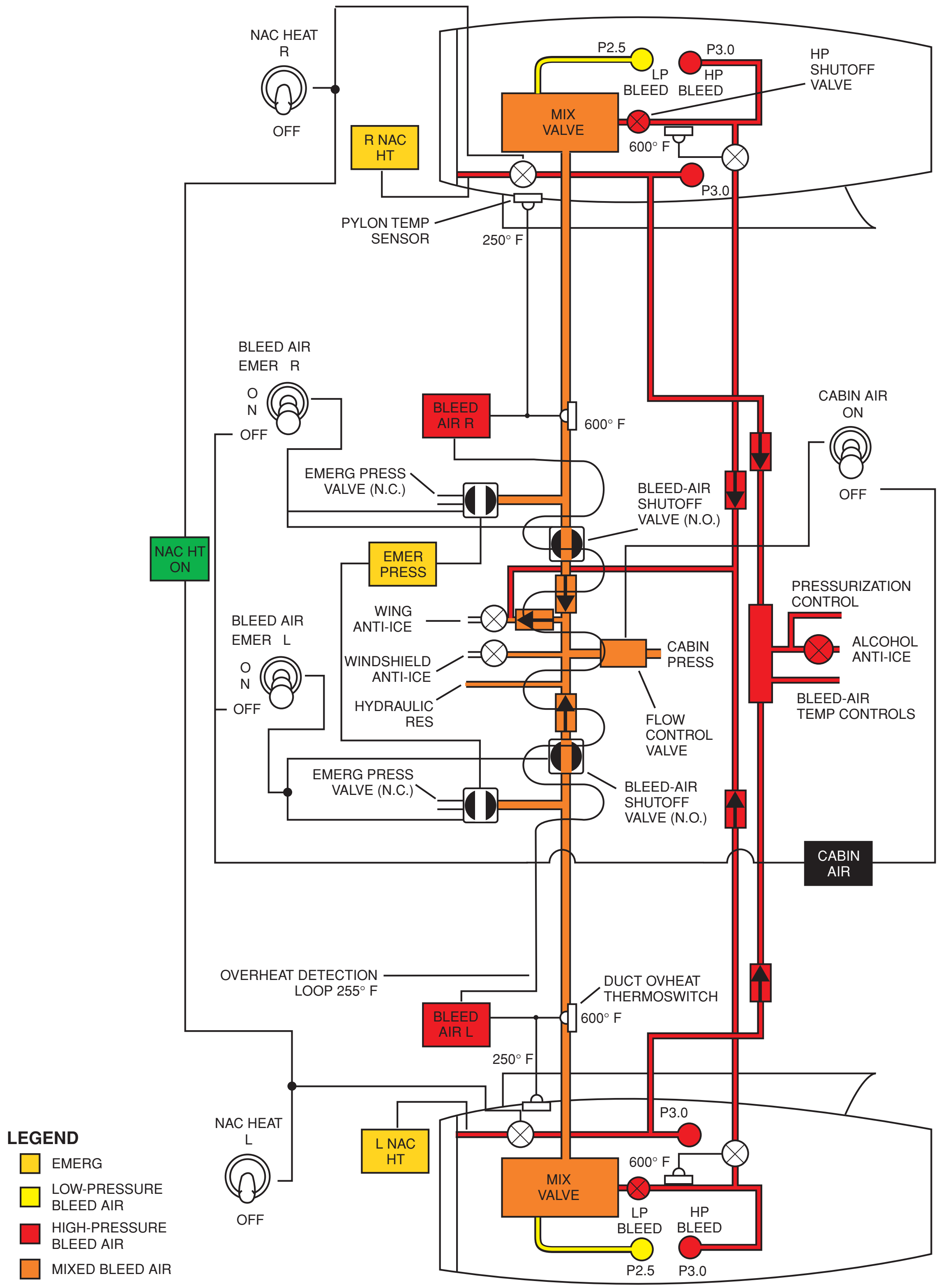

Figure 9-1. Bleed-Air System

DESCRIPTION AND OPERATION

LOW-PRESSURE (LP) BLEED-AIR SYSTEM

The LP bleed-air systems on each engine are connected through ducting in the tailcone section, forming a common manifold. The left and right engine LP system cockpit controls and indicators include:

- L and R BLEED-AIR switches
- L and R BLEED-AIR warning lights
- EMER PRESS caution light
- CABIN AIR advisory light

The LP manifold provides bleed air for cabin pressurization and temperature control, wing anti-ice, windshield anti-ice, and hydraulic reservoir pressurization.

HIGH-PRESSURE (HP) BLEED-AIR SYSTEMS

HP bleed air is taken from each engine and is ducted to two manifolds in the tailcone and to the valves necessary to operate the systems.

HP air is available to each mixing valve through the HP shutoff valve, and through a separate HP solenoid valve to the wing anti-ice system.

One of the HP manifolds provides bleed air for the engine anti-ice system, the pressurization jet pump, windshield alcohol anti-ice, and the temperature control system. The other manifold automatically provides HP bleed air to the wing anti-icing system on an as needed basis.

HP bleed air is available to the manifolds from one or both engines. The ducts contain one-way check valves to prevent reverse flow, if one engine fails. There are no cockpit control switches.

COMPONENTS

BLEED-AIR SWITCHES

The left and right LP bleed-air systems can be independently controlled with the L and R bleed-air switches (Figure 9-2) located to the right of the pressurization controller. The switches are the lever-locking type and have three positions: OFF (down), ON (center), and EMER (up).

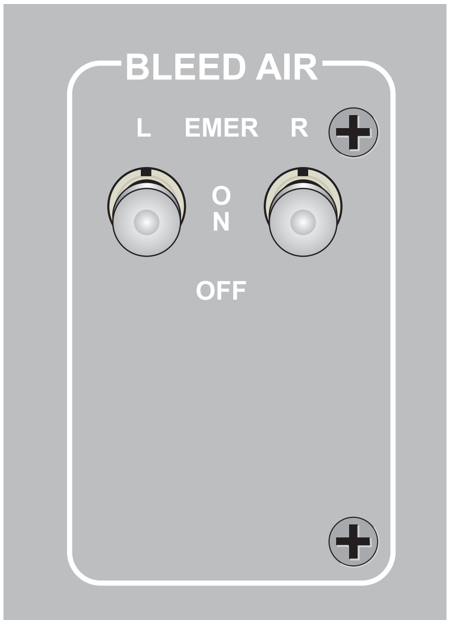

Figure 9-2. Bleed-Air Switches

MIX VALVES AND REGULATOR

A mix valve/regulator on each engine adds HP bleed air to the normal LP bleed-air manifold when needed to maintain normal operational pressure. A HP shutoff valve is normally open unless emergency pressurization has been activated. This HP air is also shut off by a poppet valve in the mixing valve system when OFF is selected on a bleed-air switch.

Either engine can supply adequate bleed-air pressure to the manifold in order to support systems operation and to maintain cabin pressure in the event of termination of the bleed-air supply from one engine due to engine or component failure.

DUCT, PYLON, AND TAILCONE TEMPERATURE SENSING

The duct overheat thermoswitches are installed inside the LP ducts where the ducts pass through the engine pylons. The duct overheat thermoswitches are set to close a circuit and illuminate the appropriate red L or R bleed-air warning light (Figure 9-1) if the temperature reaches approximately 600°F in the duct.

The pylon temperature sensors are located inside the pylon structure (Figure 9-1). They close a circuit that also illuminates the appropriate L or R bleed-air warning light if the pylon temperature reaches approximately 250°F.

A bleed-air overheat sensor in the tailcone consists of three temperature sensing loops that parallel the bleed-air lines in the fuselage and encircle the bleed-air couplings. The loop functions like the fire detect loop on the engine, both the L and R bleed-air warning lights illuminate when temperature adjacent to the loop reaches approximately 255°F. The circuit continuity is tested during the engine FIRE DET test. Proper indication is illumination of both the L and R bleed-air lights.

BLEED-AIR/SHUTOFF VALVES

The bleed-air shutoff valves are in the tailcone section. The valves are not used to regulate pressure. They are used only as shutoff valves. The valves are energized closed when the respective bleed-air switches are in the OFF or EMER position. The valves are depowered open when the bleed-air switches are in the ON position or in the event of electrical failure.

Electrical power to operate the bleed-air shutoff valves comes from L and R bleed-air circuit breakers located in the ENVIRONMENT group of circuit breakers.

In addition to being controlled with the bleed-air switches, there are two other conditions in which the shutoff valves are energized closed. These conditions are:

1. When emergency pressurization is automatically activated (aneroids)
2. When the respective ENG FIRE PULL T-handle is pulled

Emergency pressurization is addressed in the following paragraphs. The ENG FIRE PULL T-handle is explained in Chapter 8—“Fire Protection.”

EMERGENCY PRESSURIZATION

The emergency pressurization valves in the tailcone section are normally closed. They are solenoid-controlled valves and can be energized open by placing the bleed-air switches to the EMER position.

When the emergency pressurization valves are energized open, the HP shutoff valves on the engines and the bleed-air shutoff valves in the tailcone are energized closed.

With the bleed-air switches ON, emergency pressurization can be automatically activated by two (one to left side valves, one to right side valves) independent aneroid switches sensing cabin altitude. The aneroids will activate emergency pressurization at 9,500 feet or 14,500 feet, depending upon pressurization controller (landing altitude) system setting. The 9,500-foot aneroids will normally activate emergency pressurization whenever the cabin altitude reaches 9,500 feet; however, when the following three conditions exist, emergency pressurization will not occur until the cabin reaches 14,500 feet: (1) pressurization controller set to a landing field elevation above 8,000 feet, (2) aircraft has descended at least 1,000 feet from cruise altitude, and (3) aircraft is below 25,000 feet.

NOTE

Emergency pressurization cannot be activated by the aneroid switches if the bleed-air switches are in the OFF position or electrical power is not available.

With the bleed-air shutoff valves closed and the emergency pressurization valves open, low-pressure bleed air is routed directly to the cabin area, bypassing the flow control valve, heat exchanger, temperature control valves, and the ducting in the tailcone. This increases bleed-air flow into the cabin, thus restoring cabin pressurization if the pressure loss was due to an obstruction or leak in the tailcone bleed-air ducting. Since the heat exchanger is bypassed in emergency pressurization, the HP shutoff valve is energized closed to reduce the heat in the bleed-air supply.

NOTE

Crew and cabin temperature controls are ineffective in the emergency pressurization mode and the cabin and cockpit will get hot.

If cabin pressure is restored, or if a descent is made, the 9,500-foot low aneroids reset when the cabin altitude goes below 8,300 feet; the 14,500-foot high aneroids reset when the cabin altitude goes below 13,300 feet. However, the bleed-air switches must be cycled to OFF and then to the ON position to deactivate the emergency pressurization mode.

Electrical power to energize the emergency pressurization valves open and to close the bleed-air shutoff valves and HP shutoff valves is 28 VDC supplied through the L and R BLEED AIR circuit breakers.

EMER PRESS LIGHT

Whenever either emergency pressurization valve is in the open (emergency) position, whether placed there automatically by the aneroid switches or by selecting EMER position with the bleed-air switches, an amber EMER PRESS light on the glareshield panel illuminates.

NOTE

Pressurization system operating procedures are listed in Section II of the *Airplane Flight Manual* for landings at field elevations above 8,000 feet.

CABIN AIR LIGHT

In aircraft 60-271 and subsequent and prior aircraft modified by SB 60-31-1, a white CABIN AIR advisory light indicates that either L BLEED AIR, R BLEED AIR, or CABIN AIR switches are in the OFF position.

BLEED-AIR SWITCHES

OPERATION

Table 9-1 summarizes the bleed-air switch functions by switch position, assuming normal conditions. Only one switch is addressed; the same functions are applicable for the other switch.

Table 9-1. BLEED-AIR SWITCH FUNCTIONS

POSITION	EVENTS
EMER	• Bleed-air shutoff valve is energized closed. • Emergency pressurization valve is energized open. • HP shutoff valve is energized closed.
ON	• Bleed-air shutoff valve is deenergized open. • Emergency pressurization valve is deenergized closed. • HP shutoff valve is deenergized open.
OFF	• Bleed-air shutoff valve is energized closed. • Emergency pressurization valve is deenergized closed. • HP air is shut off by the poppet valve (inside mixing valve) while the HP shutoff valve is deenergized open.

NOTE

In the event of cabin pressurization loss, the 9,500-foot low aneroid switches or 14,500-foot high aneroid switches will automatically activate the emergency pressurization mode. The emergency pressurization valves and bleed-air shutoff valves will reposition as if the EMER position were selected with the bleed-air switches.

NOTE

In the event of aircraft electrical failure, the HP shutoff valves are deenergized open. Also, the emergency pressurization valves are disabled and closed (if opened) and the shutoff valves remain open, or open if previously closed.

NOTE

In emergency pressurization, HP bleed air to nacelle heat, alcohol anti-ice system, pressurization jet pump, and bleed-air temperature control valves are still available with the HP shutoff valve closed. However, bleed-air temperature control will not be available since the bleed-air heat exchanger is bypassed.

CHECK VALVES

Bleed-air check valves in the ducts (LP and HP manifolds) prevent reverse flow in the event only one engine is operating, or duct rupture occurs between the valve and the engine.

LP BLEED-AIR MANIFOLD

The LP manifold is fed by the LP bleed-air ducts from the L and R engine. The LP manifold provides bleed air for the following systems:

- Wing anti-ice
- Windshield anti-ice
- Hydraulic reservoir pressurization
- Normal cabin pressurization

NOTE

Bleed air is not available to any of the above systems when in the emergency pressurization mode. Bleed-air shutoff valves and the wing heat 600° valves close causing LP bleed to be diverted directly to the cabin.

The pressurization control module will still function properly although cabin pressurization air is supplied through the emergency pressurization valves.

FLOW CONTROL VALVE

The flow control valve (see Figure 9-1) is operated by the CAB AIR switch (Figure 9-3). If the OFF position is selected, the flow control valve, which is normally open for flight, is energized closed. This shuts off bleed air used for normal pressurization (see Chapter 11—"Air Conditioning").

In the ON position, the flow control valve is deenergized open, allowing bleed air into the pressurization and ventilation system. Part of this bleed air is cooled by routing it through the heat exchanger (in the tail) and then mixing it with hot bleed air to obtain the desired bleed-air temperature for cabin pressurization (see Chapter 11—"Air Conditioning").

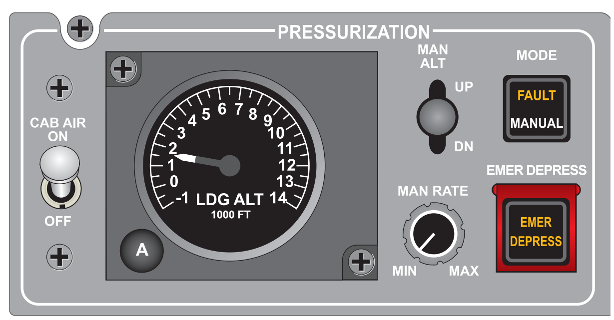

Figure 9-3. Cabin Air Switch

QUESTIONS

1. What does illumination of the EMER PRESS light indicate?
   A. BLEED-AIR switches are in the ON position.
   B. BLEED-AIR switches are in the OFF position.
   C. Emergency pressurization valves are closed and normal cabin airflow is in progress.
   D. One or both emergency pressurization valves are open.

2. Which of the following switch/controller settings would prevent automatic activation of emergency pressurization at a 9,500-foot cabin altitude?
   A. BLEED-AIR switches are OFF.
   B. Pressurization controller set to a landing altitude of 8,000 feet or less.
   C. Pressurization controller set to a landing altitude above 8,000 feet.
   D. Both A and C are correct.

3. What switch controls the flow control valve?
   A. CAB AIR switch
   B. BLEED-AIR switch
   C. Mode selector switch
   D. MAN ALT UP DN switch

4. The ___________ monitors the temperature of the bleed air in the LP ducting between the engine and the LP manifold in the tailcone section.
   A. Pylon overheat thermostat
   B. Bleed-air overheat detection loop in the tailcone
   C. Pylon temperature thermoswitch
   D. Bleed-air duct overheat thermoswitch

5. When will the L BLEED-AIR light illuminate?
   A. When the temperature in the left pylon or the left bleed-air duct exceeds its predetermined limit.
   B. When the pressure in the left pylon is below the system's operational limit.
   C. When the left half of the bleed-air system is operating.
   D. When the left half of the bleed-air system has failed.

6. When will both the BLEED-AIR L and R lights illuminate?
   A. L and R BLEED-AIR switches are OFF.
   B. Tailcone overheat sensor exceeded predetermined limit.
   C. FIRE DET test is in progress.
   D. B and C are both correct.

7. With emergency pressurization activated, which of the following systems is (are) inoperative?
   A. Windshield and wing anti-ice (bleed air)
   B. Nacelle anti-ice heat (bleed air)
   C. Alcohol anti-ice
   D. Pressurization jet pump

CHAPTER 10
ICE AND RAIN PROTECTION

CONTENTS

ILLUSTRATIONS

CHAPTER 10
ICE AND RAIN PROTECTION

INTRODUCTION

Anti-icing equipment on the Learjet 60 is designed to prevent buildup of ice on:

- The engine nacelle lip, inner stator vanes, engine inlet pressure and temperature probes, and engine fan spinner
- The windshield
- The leading edges of the lifting surfaces (stabilizer and wings)
- The pitot-static probes, stall warning vanes, pressurization static port and total temperature probe

GENERAL

All anti-icing equipment must be turned on before icing conditions are encountered. To delay until ice buildup is visually detected on aircraft surfaces constitutes an unacceptable hazard to safety of flight.

Icing conditions exist when the SAT in flight is 10°C (50°F) to –40°C (–40°F), and visible moisture in any form is present (such as clouds, rain, snow, sleet, ice crystals, or fog with visibility of one mile or less). Nacelle heat must be on during descent into icing conditions even if the SAT is below –40°C (–40°F).

Icing conditions also exist when the SAT on the ground and for takeoff is 10°C (50°F) or

below when operating on ramps, taxiways, or runways where surface snow, ice, standing water, or slush may be ingested by the engines, or freeze on engines nacelles, or engine sensor probes.

ICE DETECTION

During daylight operations, ice accumulation can be visually detected on the lower corners of the windshield and on the wing leading edge.

WINDSHIELD ICE DETECTION

During night operation, the windshield ice detection lights indicate ice or moisture formation on the windshield. Two tubes, one on the pilot side of the glareshield and one on the copilot side, contain red lights which continuously shine on the inside of the windshield. The ice detection lights normally shine through unseen. However, they will reflect red spots approximately 1 1/2 inches in diameter if ice or moisture has formed on the windshield.

The ice detection light (Figure 10-1) on the pilot side is located inside the anti-ice airstream; the light on the copilot side is located outside the anti-ice airstream. For this reason, the copilot light must be monitored whenever windshield heat or the alcohol anti-ice systems are operating. The lights indicate ice encounters when SAT is below freezing, and moisture encounters when SAT is above freezing.

The lights are powered by DC current through the L and R ICE DETECT LIGHT circuit breakers on the pilot and copilot ANTI-ICE group of circuit breakers.

WING ICE DETECTION

The wing inspection light may be used to visually inspect the leading edge of the right wing and winglet for ice accumulation during night operations. The light is illuminated by depressing the WING INSP LIGHT button on the copilot R INSTR LIGHTS panel. The light is focused on a black spot on the outboard wing leading edge to enhance visual detection of ice accumulation.

Power is supplied through the WING INSP LT circuit breaker on the copilot LIGHTS circuit-breakers group.

NOTE

Clear ice on the wing leading edge may not be detectable from the cockpit.

The wing structure can be monitored for possible icing conditions in flight by using the WING TEMP indicator (Figure 10-2). The temperature indication is blue-green-red; temperature at the blue-green line is approximately 80°F (26.7°C), and at the green-red line, approximately 190°F (88.0°C).

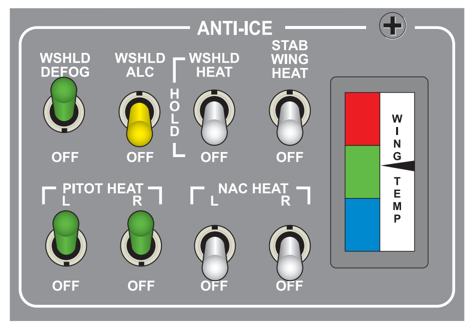

Figure 10-2. Anti-ice Control Panel

ANTI-ICE SYSTEMS

Aircraft anti-icing is accomplished through the use of electrically heated anti-ice systems, engine bleed-air heated anti-ice systems, and an alcohol anti-ice system.

Electrically heated systems include pitot-static probes, engine inlet air pressure-temperature sensor, stall warning vanes, horizontal stabilizer, total temperature probe, interior windshield defog, and pressurization static port.

Engine bleed air is used to heat the windshields, wing leading edges, nacelle inlets, engine inlet inner stators, and engine fan spinners.

An alcohol anti-ice system is installed as a backup for the pilot windshield bleed-air anti-ice system.

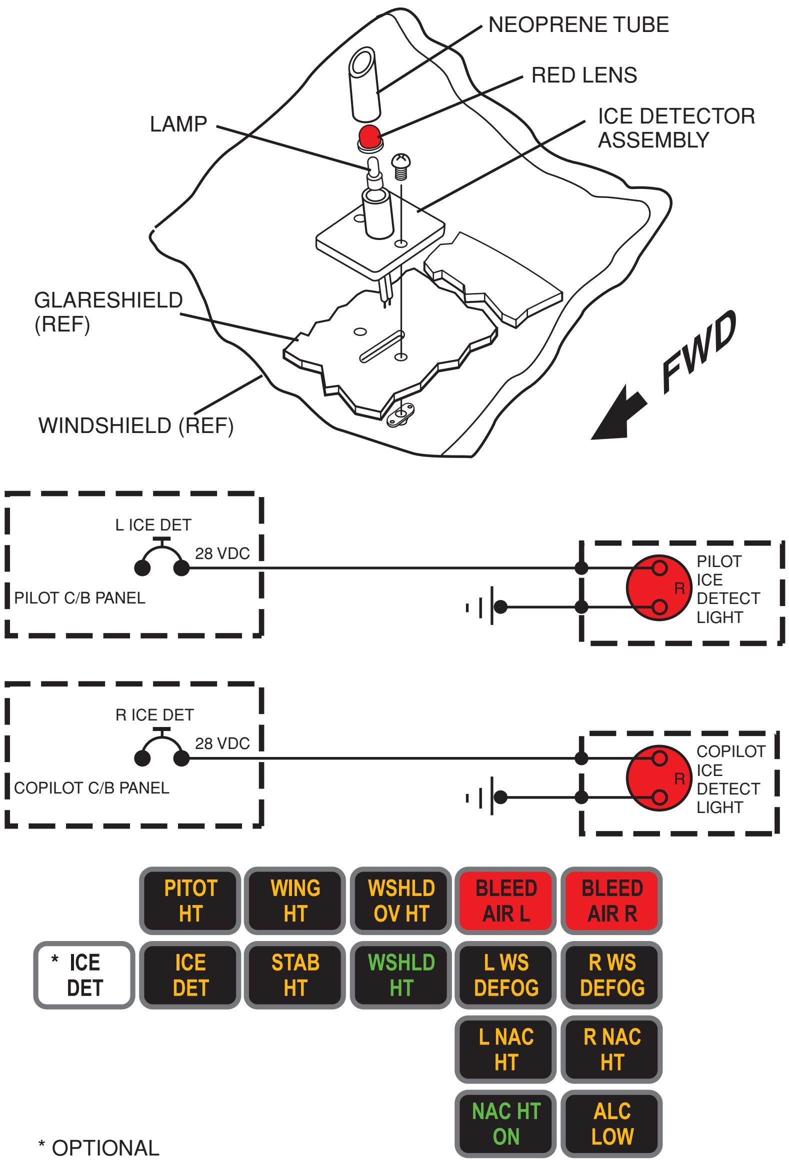

Figure 10-1. Ice Detect Lights and All Glareshield Anti-ice Lights

NOTE

All anti-ice systems require electrical power to operate except the engine inlet inner stator vanes and nacelle lip bleed-air heating system which fail ON when electrical power is not available to their respective anti-icing valves. The engine fan spinner is heated continuously by bleed air from inside the engine as long as the engine is operating.

If any anti-ice system fails, its circuit breaker should be checked and reset if necessary.

The anti-ice systems must be turned on before icing conditions are encountered.

If anti-ice systems are required during takeoff, they should be turned on just prior to setting takeoff power. Appropriate takeoff power and performance charts must be used.

When using anti-ice systems, maintain sufficient engine rpm to keep the WING TEMP indicator in the green band. This will provide sufficient bleed air for all bleed-air anti-icing systems.

When anti-ice systems are used at high altitudes, the cabin altitude may increase unless engine rpm is maintained to compensate for the additional bleed-air use.

See the *Airplane Flight Manual* for limitations and normal procedures related to the anti-icing equipment. In the event of a malfunction of any anti-icing equipment, see procedures listed under Anti-Icing in the Abnormal Procedures section of the *AFM*. The Pitot-Static System Malfunction is listed under Instruments/Avionics in the Abnormal Procedures section of the *AFM*.

ENGINE ANTI-ICE SYSTEM

The engine anti-ice system provides anti-ice protection for the engine nacelle inlet lips, the engine inlet stators, engine fan spinners, and the engine inlet air-pressure and temperature sensor heaters (P_1T_1 probes).

The engine nacelle lips, engine inlet inner stators, and engine fan spinners are heated with bleed air. The engine inlet air-pressure and temperature sensor heaters are electrically heated.

Nacelle Heat Switches

Each engine anti-ice system is independently controlled by NAC HEAT L and R switches located on the ANTI-ICE control panel (see Figure 10-2).

Each NAC HEAT switch has two positions: ON (L or R) and OFF. In the ON (L or R) position, the electrical elements in the associated air-pressure and temperature (P_1T_1) sensors are energized, and the associated nacelle heat shutoff and engine anti-ice solenoid valves are open (Figure 10-3). When either or both nacelle heat switches are on, the green NAC HT ON light is illuminated on the annunciator panel (see Figure 10-1). The associated N_1 bug will also indicate a lower thrust rating.

Electrical power to operate the nacelle heat systems is DC voltage, provided through the L and R NAC HT circuit breakers on the pilot and copilot ANTI-ICE circuit-breaker group.

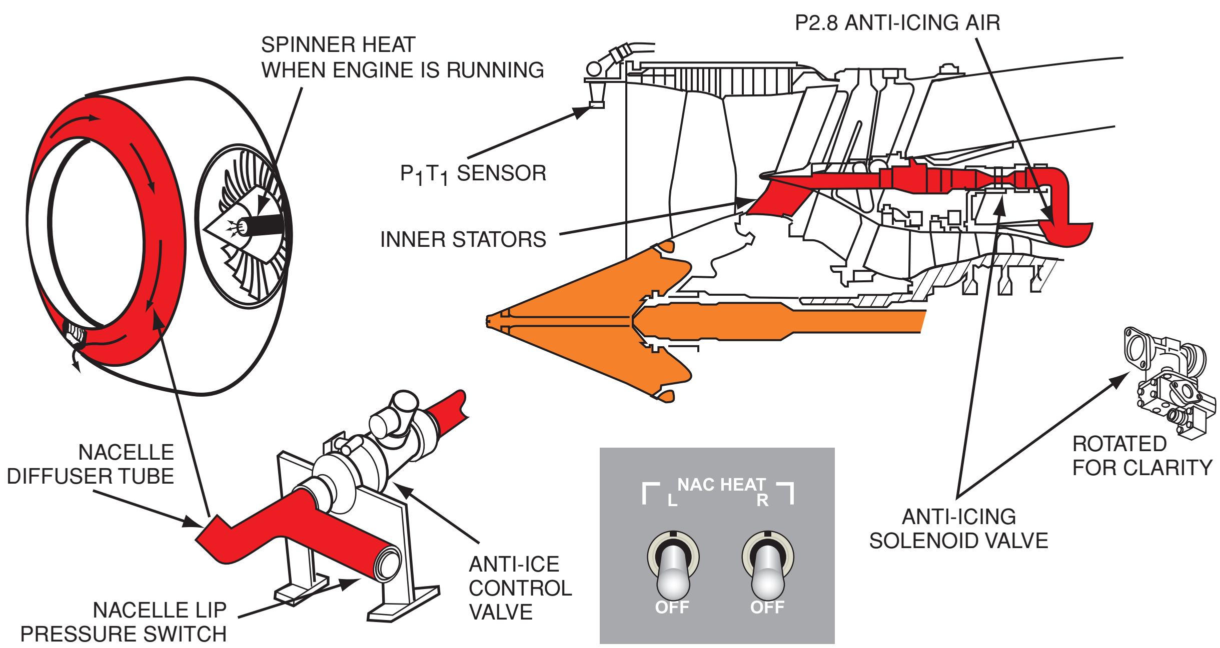

Figure 10-3. Nacelle and Engine Heat Schematic

10 ICE AND RAIN PROTECTION

NOTE

The nacelle lip anti-ice and the engine inlet inner stator vanes anti-ice fail ON with the loss of electrical power.

Bleed air for nacelle lip anti-icing is taken from the inboard HP (P3.0) bleed-air position on the engine. It is ducted through the nacelle heat shutoff valve to a swirl nozzle which distributes it to the inner surface of the nacelle lip. The air is then exhausted overboard through a vent at the bottom of the nacelle lip.

Bleed air for the engine anti-ice of the inner fan stators of the core engine is taken from LP (P2.8) bleed-air position on the engine. It is ducted through a transfer tube from the anti-icing solenoid valve to a cavity in the intermediate case and then to the inner fan stators. Air exiting from the fan stators is vented to the core airflow.

Bleed air for the engine fan spinners (engine nose cone) is taken from the LP (P2.8) bleed-air position on the engine.

It is first ducted through the intermediate case of the engine, then through the inside of the LP shaft, through the hollow nose cone extension, and into the double-walled nose cone. Air exits from the nose cone, through the nose central tie bolt and the rear of the nose cone, into the inlet airstream ahead of the fan (LP compressor).

Engine Ice Lights

The amber L and R NAC HT light on the glareshield annunciator panel provides the crew with visual indication of nacelle lip anti-ice system malfunction. The light is operated by a pressure switch in the associated nacelle lip bleed-air plumbing and a proximity switch built into the anti-icing solenoid valve. These lights are disagreement lights. Illumination of a L or R NAC HT light in the ON position indicates that bleed-air pressure to the nacelle lip is not sufficient to provide satisfactory anti-ice protection or the anti-ice solenoid valve has failed to open.

Illumination of either the L or R NAC HT amber light, with the associated NAC HEAT switch OFF, indicates that nacelle heat (bleed-air pressure) is still being applied to the nacelle lip heat system and innerstator heat system due to a malfunction of the nacelle anti-ice control valve or innerstator anti-icing solenoid valve.

Cycling the NAC HEAT ON and back to OFF may close the open valve. Also, pulling the appropriate L or R NAC HT circuit breaker will depower the control valve and should cause it to open.

The green NAC HT ON light illuminates anytime a NAC HEAT switch is set to ON (L or R).

The respective N_1 bug will drop on the appropriate engine to indicate the lower anti-ice on power rating.

Ground limitations on nacelle heat can be found in the Limitations section of the *AFM*.

WINDSHIELD ANTI-ICE/DEFOG SYSTEMS

Exterior Windshield Anti-ice/ Defog/Rain Removal System

The WSHLD HEAT switch controls flow of engine bleed air to the exterior of the windshield for anti-icing and to help defogging. The three-position switch is labeled WSHLD HEAT–HOLD–OFF and is located on the ANTI-ICE control panel (see Figure 10-2).

Engine bleed air from the bleed-air manifold is routed through two valves: the windshield anti-ice shutoff valve (open or closed) and the windshield anti-ice modulating valve (Figure 10-4). The shutoff valve is spring-loaded closed and powered open whenever aircraft electrical power is on and the WSHLD HEAT circuit breaker, on the copilot ANTI-ICE group of circuit breakers, is in.

In the event of electrical failure, windshield heat will be inoperative. Even if the modulating valve is open at the time of failure, the shutoff valve will close.

During normal operation, when the WSHLD HEAT switch is positioned to WSHLD HEAT, the green WSHLD HT light, on the glareshield annunciator panel, illuminates and the anti-ice modulating valve drives to full open within approximately 15 seconds. It also takes 15 seconds to fully close this valve once off is selected.

For reduced airflow to the windshield, the modulating valve may be stopped in an intermediate position (between full open or full closed) by positioning the WSHLD HEAT switch to HOLD when the modulating valve is in the desired intermediate position.

With the WSHLD HEAT switch positioned to WSHLD HEAT, bleed air flows through the shutoff valve and the anti-ice modulating valve into the heat exchanger and is then ducted to the outlet nozzles at the base of the windshield. The anti-ice heat exchanger cools the bleed air with ram air, regulated by a ram-air modulating valve. This valve is controlled by an anti-ice duct temperature sensor 300°F (149°C) that regulates the anti-ice bleed-air temperature by varying the amount of ram air allowed into the heat exchanger. During ground operation, ram air is not available to cool the bleed air (Figure 10-4).

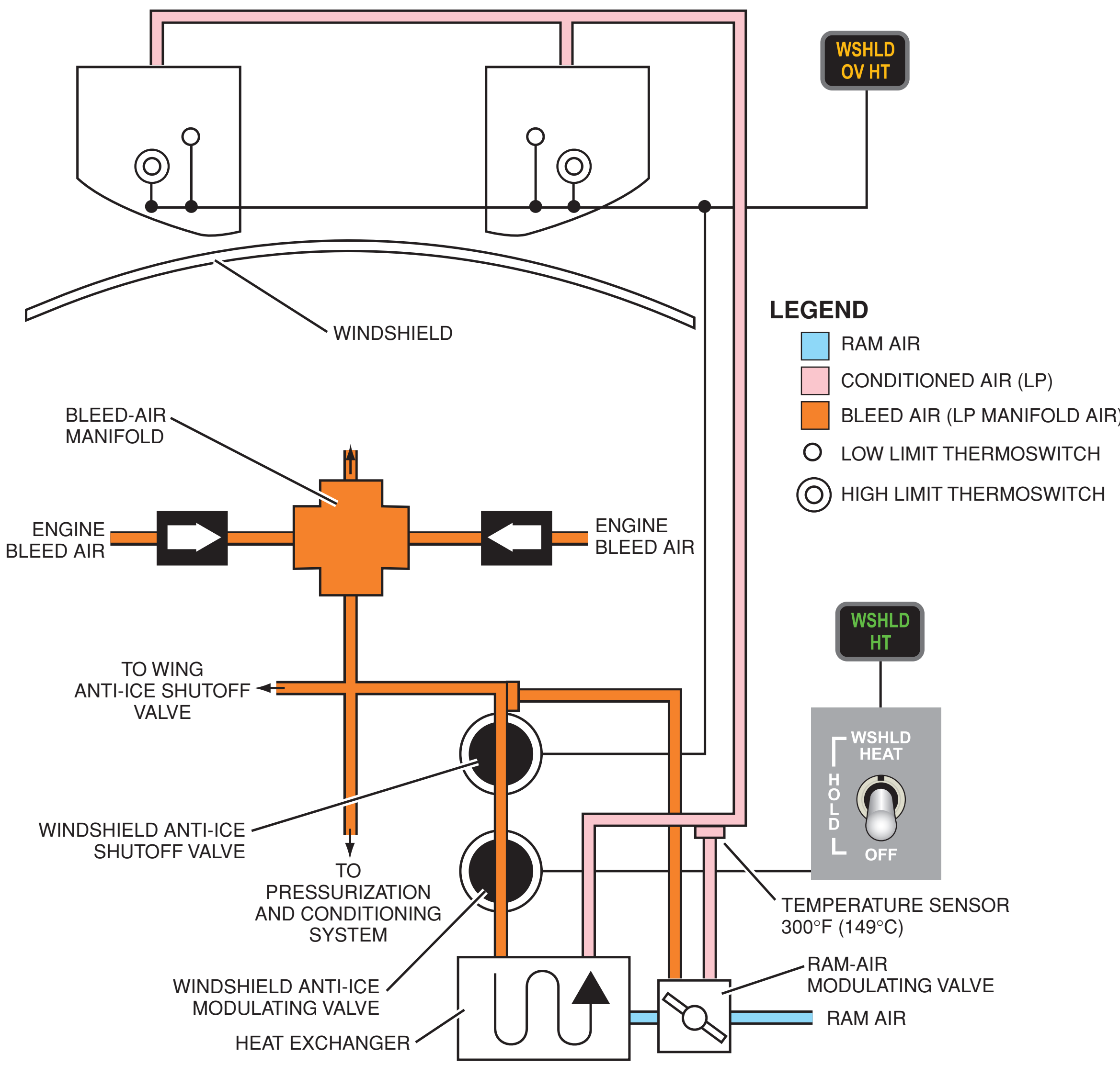

Figure 10-4. Windshield Anti-ice System

Under normal in-flight conditions, the windshield heat bleed-air temperature is automatically controlled. However, an overheat warning system is installed to alert the pilot and automatically shut off windshield heat, in the event of an overheat condition. There is a low-limit (approximately 250°F) and a high-limit (approximately 347°F) thermoswitch installed in each windshield. The low-limit thermoswitches function only on the ground and are cut out by the squat switch relay box when airborne. The high-limit thermoswitches are installed primarily to limit temperature during airborne operation, but will also function on the ground as a backup to the low-limit thermoswitches.

If either outlet nozzle temperature reaches 250°F limit (ground) or 347°F limit (airborne), the tripped thermoswitch will illuminate the amber WSHLD OVHT, on the glareshield annunciator panel, and cause the shutoff valve to close. The anti-ice modulating valve will remain in the position it was in, but the green WSHLD HT light will be extinguished while the shutoff valve is closed. The WSHLD OVHT light will extinguish and the shutoff valve will open when the nozzle temperature drops to approximately 240°F (ground) or approximately 311°F (airborne). If the WSHLD HEAT is not turned OFF after the WSHLD OVHT light illuminates, airflow will resume to the windshield, and the green WSHLD HT light will illuminate. To avoid a false WSHLD OVHT indication on landing, the low-limit overheat thermoswitch circuitry is disabled for 10 seconds after touchdown, after which normal functioning will resume.

NOTE

Bleed air is not available for windshield anti-icing with both the left and right bleed-air systems in the emergency pressurization mode (bleed air shutoff valves are closed).

Alcohol Anti-ice System

The alcohol anti-icing system is provided as a backup for the normal exterior windshield anti-ice system and is available for use on the pilot windshield only.

Alcohol anti-icing is accomplished by directing methyl alcohol (methanol) over the pilot windshield surface through an external outlet in the left windshield heat outlet nozzle assembly.

The system has a 2.35 gallon reservoir located in the left forward avionics bay which will provide for 45 minutes of alcohol anti-ice operation when fully serviced. The reservoir contains a float switch which illuminates an amber ALC LOW caution light on the glareshield annunciator panel when the reservoir is empty.

The alcohol anti-ice system is controlled by the WSHLD ALC switch on the ANTI-ICE control panel (see Figure 10-2). The switch has two positions: WSHLD ALC and OFF.

When the switch is set to WSHLD ALC, circuits are completed to open the shutoff and pressure regulator valve and the control valve (Figure 10-5). When the shutoff and regulator valve is open, regulated bleed air pressurizes the reservoir, forcing the fluid through the control valve to the windshield.

NOTE

Alcohol anti-ice operation is not affected by the emergency pressurization mode since bleed air for alcohol anti-icing comes from a different (HP) bleed-air manifold in the tailcone.

The alcohol anti-ice system control circuits operate on DC current supplied through the ALCOHOL SYSTEM circuit breaker in the copilot ANTI-ICE group of circuit breakers.

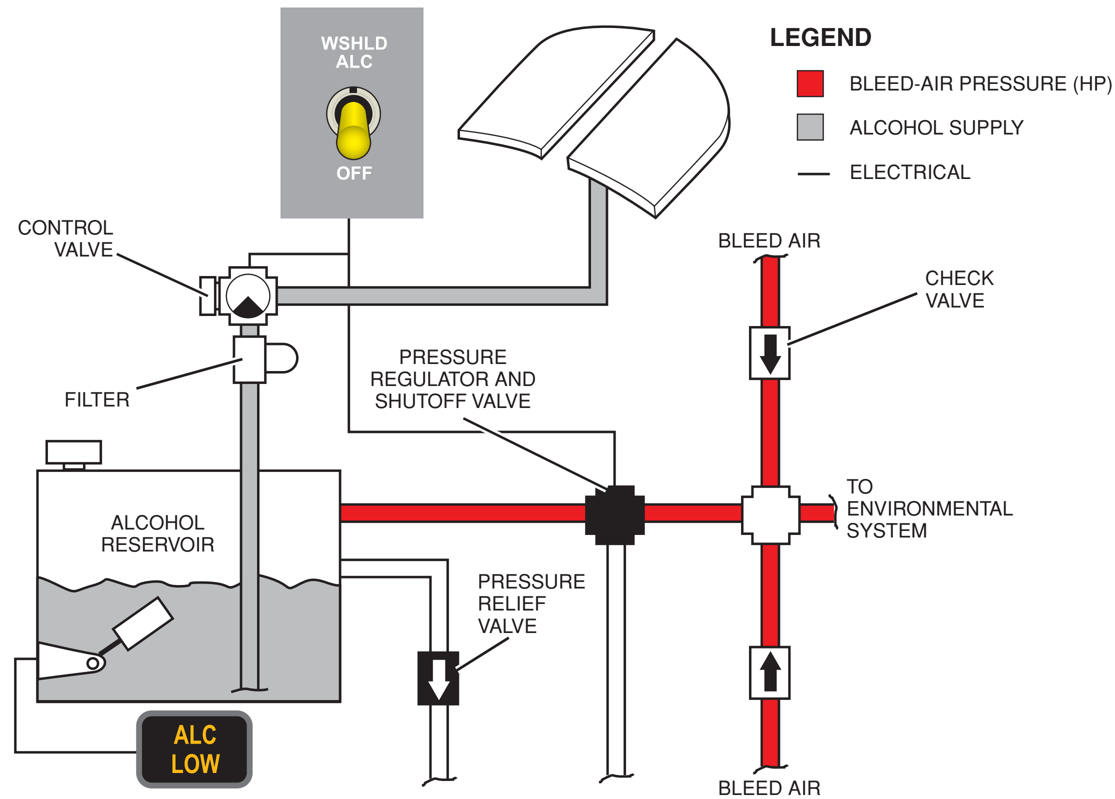

Figure 10-5. Windshield Alcohol Anti-ice System Schematic

Interior Windshield Defog System

General

Windshield interior defogging (Figure 10-6) is provided by electrically heated windshield panels. Components of the system are: two windshield heaters (integral with windshield), four heat sensors, a windshield defog relay box, a WSHLD DEFOG–OFF switch, two (L WS DEFOG and R WS DEFOG) overheat/underheat annunciators, four (two L WSHLD DEFOG and two R WSHLD DEFOG) circuit breakers, and an inverter relay box. Power for the windshield electric defog heaters is supplied by the left and right inverters (Figure 10-6).

Operation

When the WSHLD DEFOG–OFF switch is positioned to WSHLD DEFOG, AC electrical power is routed through the windshield defog relay box to the heating elements on the pilot and copilot windshield.

Power to the system is confirmed by the amber L WS DEFOG and R WS DEFOG lights illuminating on the glareshield annunciator panel. When windshield temperature reaches approximately 85°F the respective L WS DEFOG/ R WS DEFOG light(s) go out. The WS DEFOG lights will not illuminate when the switch is turned on if the windshield temperature is already above 85°F. Operation can also be confirmed by touching the windshield to confirm it is warm. After the initial activation, the L WS DEFOG/R WS DEFOG light(s) are failure lights (overheat 150°F, underheat below 85°C). During normal operation, the system maintains approximately 110°F windshield temperature. Aircraft 60-114 and subsequent (or 60-001 through 60-113 incorporating SB 60-30-2) have a three-position switch labeled OFF–LOW–NORM. The LOW position will

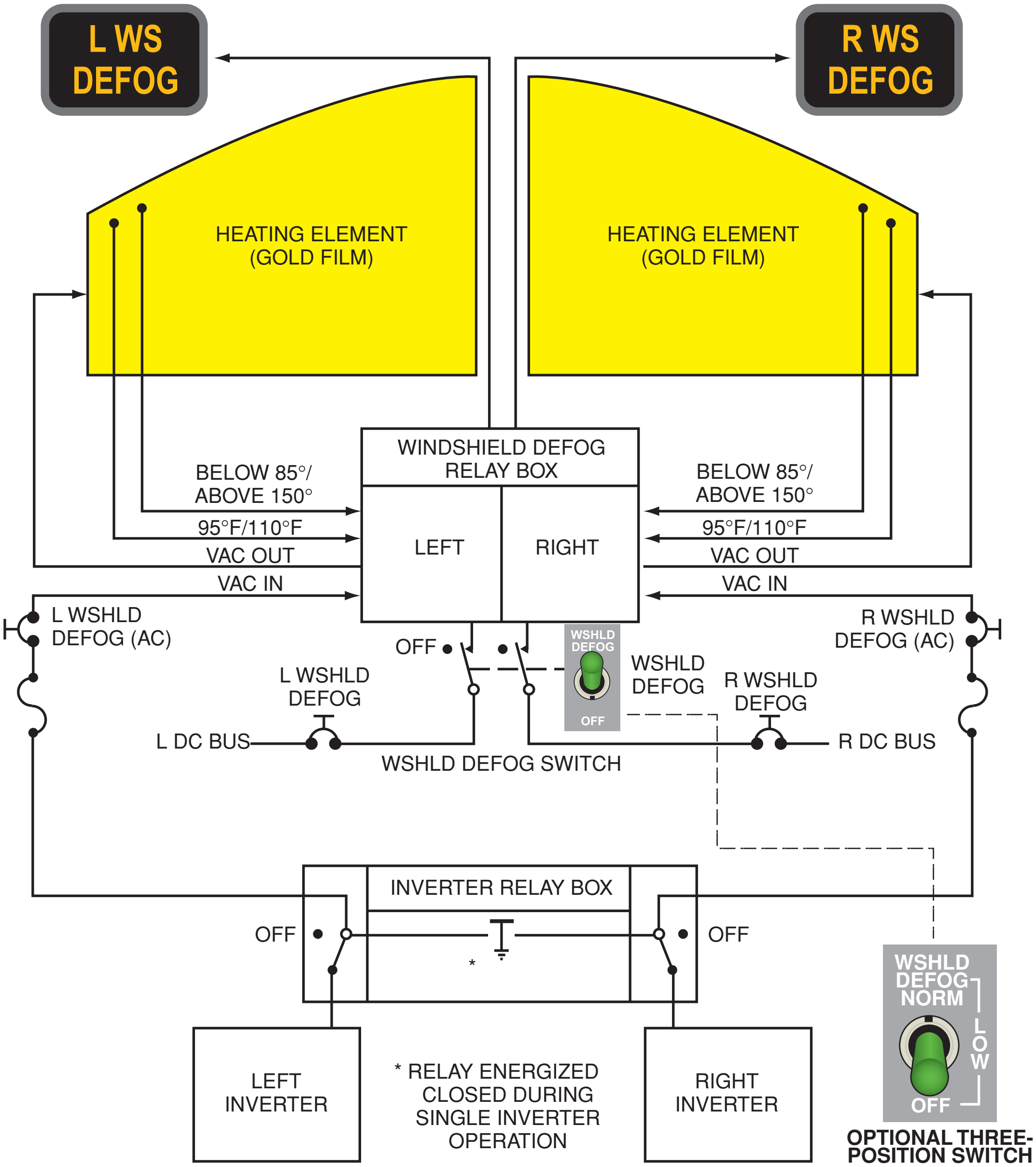

Figure 10-6. Windshield Defog System (Interior)

operate the system to maintain approximately 95°F windshield temperature. During single inverter operation, both windshield heating elements are heated by the single inverter (Figure 10-6), provided the failed inverter is turned off.

During normal two inverter operations the pilot windshield is powered by the left inverter, and the copilot by the right inverter.

WSHLD DEFOG Switch and Circuit Breakers

The WSHLD DEFOG–OFF switch is located on the ANTI-ICE switch panel (Figure 10-2). In the WSHLD DEFOG position, the electric windshield defogging is in operation. In the OFF position, the system is shut down. Electrical power is supplied by two L WSHLD DEFOG circuit breakers, (one DC and one

AC), on the pilot ANTI-ICE circuit-breaker group and two R WSHLD DEFOG circuit breakers (one DC and one AC), on the copilot ANTI-ICE circuit-breaker group (see Figure 10-6). L and R WSHLD DEFOG lights are located on the glareshield annunciator panel and indicate either windshield temperature is below approximately 85°F or above approximately 150°F, or loss of AC or DC power.

WING ANTI-ICE SYSTEM

The wing anti-ice system (Figure 10-7) utilizes engine bleed air directed through diffuser tubes in each wing leading edge. The heated air from the diffusers warms the leading edge of the wings and is then directed into the center wing/wheel well area to heat the main gear wheels and brakes.

STAB WING HEAT SWITCH (WING HEAT FUNCTION)

The wing heat system is controlled with the STAB/WING heat switch on the ANTI-ICE control panel (see Figure 10-2). Although STAB anti-ice utilizes an electric blanket, and wing anti-ice utilizes bleed air, both systems are controlled with the same switch. The switch has two positions: STAB WING HEAT and OFF.

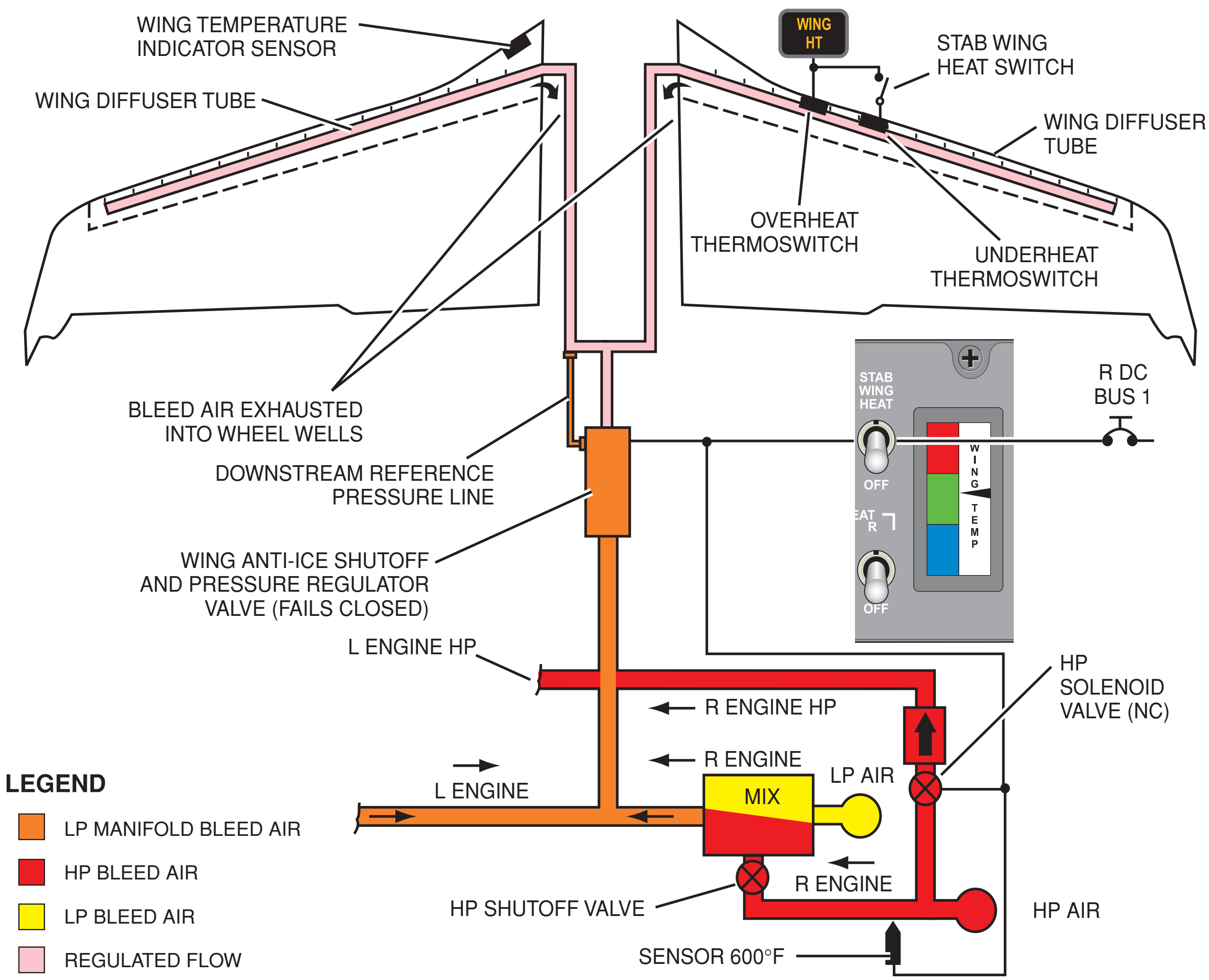

Figure 10-7. Wing Anti-ice System

In the STAB WING HEAT position, the switch energizes the wing anti-ice shutoff and pressure regulator valve and allows hot air to flow through the ducting into the wing diffuser tubes. The valve will maintain a regulated bleed airflow to the wings.

In the event of an electrical failure, the shutoff valve fails closed and wing anti-ice will not be available. When energized, the amber WING HT light will illuminate indicating underheat of the wing. As the wing heat reaches the normal operating range above 70°F, the WING HT light goes out. As long as wing heat is in the normal operating range (green range wing temperature indicator) the WING HT light is out. If the wing temperature reaches 215°F the amber WING HT light illuminates, indicating wing overheat.

If wing temperature decreases below 58°F the amber WNG HT light will illuminate, indicating wing underheat and ice may form (Figure 10-7).

With the wing switch in the STAB WING HEAT position, and HP air temperature sensor below 600°F, electrical power is applied which opens a normally closed (NC) HP solenoid (located on each engine) valve. This allows hot HP air to be routed directly to the wing heat ducting to help maintain wing heat in the operating range. This is especially appropriate as engine rpm is reduced for a descent through icing conditions.

As engine rpm is increased, and the HP air temperature sensor goes above 600°F, electrical power is removed and the HP solenoid valve closes. It is possible under certain conditions that the wing temperature sensor will actually show a decrease in temperature when engine rpm is increased. It will, however, stay in the normal operating range.

Electrical power for control of the system is supplied through the WING HEAT circuit breaker on the copilot ANTI-ICE circuit-breaker group.

Wing Temperature Sensor

Wing temperatures are sensed by a temperature sensor, installed on the inside of the left wing leading edge (Figure 10-7), and indicated on the WING TEMP indicator (see Figure 10-2), on the ANTI-ICE switch panel (bottom center of the instrument panel). If the indicator is in the blue segment, the wing leading edge-temperature is approaching an underheat condition. If the indicator is in the green segment, wing leading-edge temperature is above 80°F. If the indicator is in the red segment, the wing leading edge is approaching an overheat condition (above 190°F).

Electrical power for the temperature indicator is supplied through the WING HEAT circuit breaker on the copilot ANTI-ICE circuit-breaker group.

WING HT Light

If the wing temperature reaches approximately 215°F (102°C) a thermoswitch (Figure 10-7) in the RH wing to fuselage fairing illuminates the amber WING HT light on the glareshield annunciator indicating a wing overheat. If wing temperature decreases to approximately 58°F (14.4°C) or less, the thermoswitch illuminates the same amber WING HT light, indicating a wing underheat. Electrical power for the light is supplied through the WARN LTS circuit breaker on the pilot and copilot LIGHTS group of circuit breakers.

NOTE

The low heat function of the WING HT light functions only when the STAB WING HEAT switch is in the STAB WING HEAT position. The overheat function operates with the STAB WING HEAT ON or OFF.

HORIZONTAL STABILIZER ANTI-ICE SYSTEM

The horizontal stabilizer anti-ice system utilizes sequenced electrical current to elements embedded in two rubberized blankets (Figure 10-8), and one element along the leading edge of each half of the horizontal stabilizer. There are seven elements extending lengthwise in each blanket: a parting element (at the leading edge), three elements above, and three elements below the parting element (Figure 10-9).

Figure 10-8. Horizontal Stabilizer Heating Blanket

The system (Figure 10-9) consists of: two blankets (seven elements in each blanket), an electronic controller, a control switch, and a ground test circuit.

When circuit power is applied to the electronic controller, it performs a short test. Then it applies continuous electrical power to the two parting elements and starts sequencing circuitry that alternately distributes electrical power to the 12 other elements for 15 seconds duration each (three minutes for a full cycle). Except for a test sequence at initial turn-on, electrical power to the stab heat blanket is cut out on the ground by the left main gear squat switch.

During normal stabilizer heat operation amperage on the electric power monitor will reflect a current draw increase of approximately 60 amps per generator (120amps if on APU).

At least one generator must be operating to energize the stabilizer heat control circuits. An APU may be used for the ground test. The control circuits are also routed through a start cutout relay; therefore, the system is inoperative during starter-assisted engine starts.

NOTE

> Caution must be exercised if performing the ground self-test on GPU only because there will be no indication of amps in the cockpit. The increase of amps for three to five seconds is required for a valid ground test.

For ground operation to prevent overheating of the horizontal stabilizer heating elements, ensure that the amber STAB HEAT light is illuminated steady. On the ground, activation of the system initiates a self-test characterized by the illumination of the STAB HEAT light and an amps increase for approximately three seconds. If the STAB HEAT light is not illuminated steady and amps do not reduce within five seconds, immediately set STAB WING HEAT switch to OFF. Flashing of the STAB HEAT light (or no light) during the self-test indicates a failure in the system. Flashing of the STAB HEAT light, in flight, indicates a failed heating element (the remaining elements will continue to function normally). The test sequence requires a minimum of three minutes between each test.

Electrical power, to heat the elements, is 28 VDC, supplied from the battery charging bus through three 50 amp current limiters, three remote control circuit breakers, and an electronic controller.

The control circuit uses 28 VDC supplied through the STAB HT circuit breaker on the copilot ANTI-ICE group of circuit breakers.

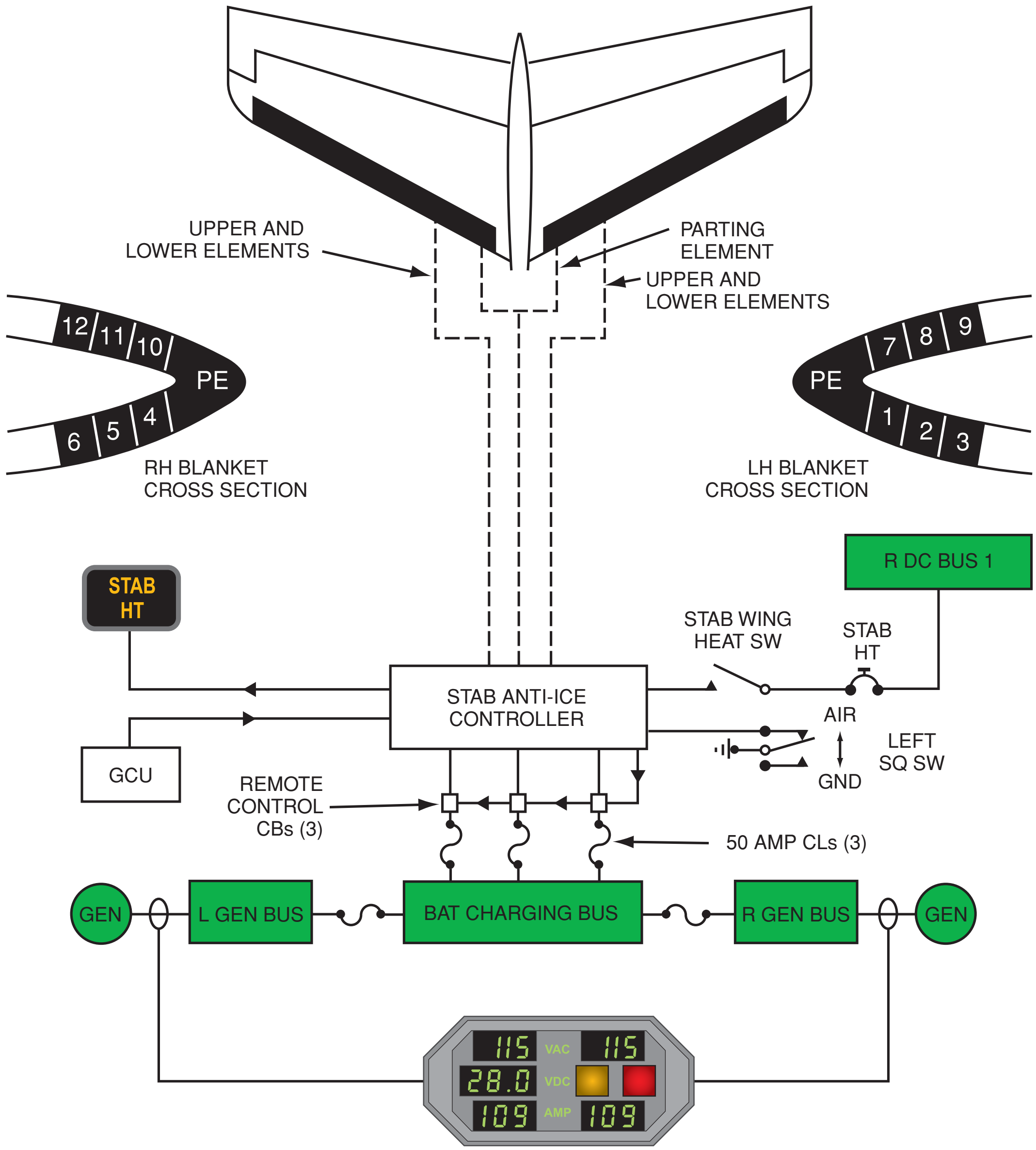

Figure 10-9. Horizontal Stabilizer Heating System

STAB WING HEAT Switch—STAB HT Function

The STAB WING HEAT switch is located on the ANTI-ICE control panel and is a two-position switch (see Figure 10-2) labeled STAB WING HEAT and OFF.

This switch controls the ground test and, in flight, provides control circuitry to operate the system.

STAB HT Light

The amber STAB HT light, located on the glareshield annunciator panel (see Figure 10-1), will illuminate whenever the switch is placed to the STAB WING HEAT position and any one element(s) (parting or sequencing element[s]) have failed or electrical power has failed. Monitoring of the amperage while cycling the STAB WING HEAT switch OFF and then ON, can be used to determine the

degree of system failure. During ground operation, the light will illuminate steady during the ground test of the system.

PITOT-STATIC AND STALL WARNING ANTI-ICE SYSTEMS

Anti-ice protection for the pitot-static probes, AOA vanes, pressurization differential static port, and total temperature probe is accomplished with integral, electrical heating elements in each component (Figure 10-10).

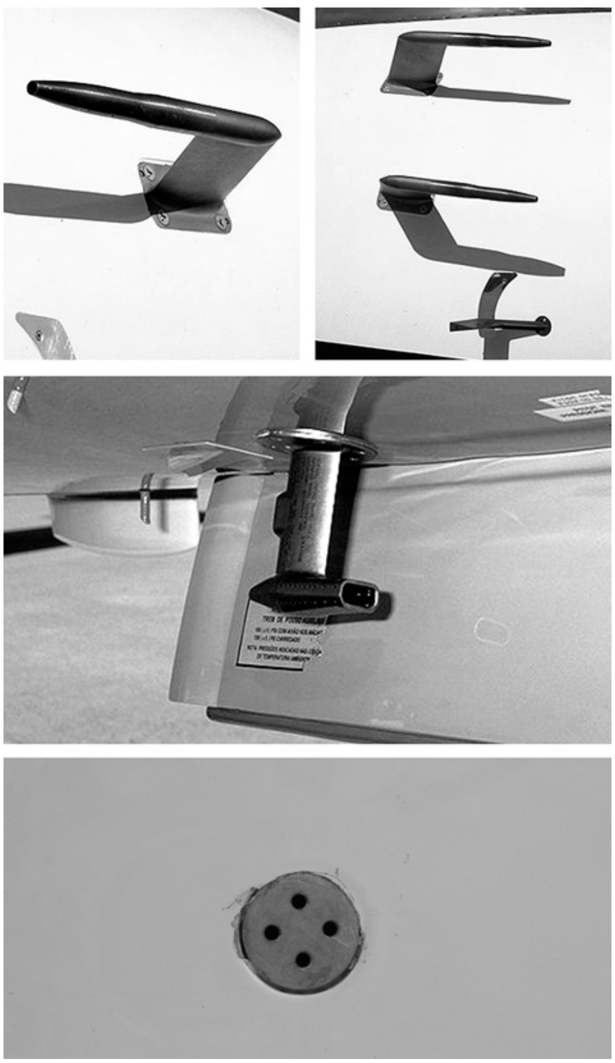

Figure 10-10. Pitot-Static Probe, AOA Transducer Vane, Pressurization System Static Port, and Total Temperature Probe

This anti-ice protection is divided into two systems (left and right) which are controlled by the L and R PITOT HEAT switches (Figure 10-11). The left pitot heat system includes the left pitot static probe and the left stall warning vane.

The right heat system includes the right pitot-static probe, standby pitot static probe, total temperature probe, pressurization differential static port, and right stall warning vane.

The left and right main pitot-static probe heating elements receive DC current through the L PITOT HEAT and R PITOT–STALL–TAT HEAT circuit breakers in the pilot and copilot ANTI-ICE circuit-breaker groups, respectively. The standby pitot-static probe heating element receives DC current through the STANDBY PITOT HEAT circuit breaker on the copilot ANTI-ICE circuit-breaker group. All three pitot-static probe heating elements normally receive power from the battery charging bus when the PITOT HEAT switches are on; however, in the emergency bus mode of operation the main (left and right) pitot-static probe heat elements will not be powered, but the standby pitot-static probe will be heated, with current from the right aircraft battery, if the right PITOT HEAT switch is on.

The heat elements in left and right stall vanes are powered from the left DC BUS 3 and right DC BUS 1 through the L STALL VANE HEAT and the R STALL VANE HEAT circuit breakers, respectively, when the left and right PITOT HEAT switches are on.

The TAT probe heat element receives power from the battery charging bus through the TAT PROBE HEAT circuit breaker in the copilot ANTI-ICE group of circuit breakers. However, before the heat element is energized, the R PITOT-STALL-TAT HEAT circuit breaker must be in, the right PITOT HEAT switch on, and the squat switch relay box must be in the airborne mode.

The heat element, in the pressurization differential static port on the right side of the aircraft nose, is energized through the right pitot-static probe heat circuit any time the right PITOT HEAT switch is on.

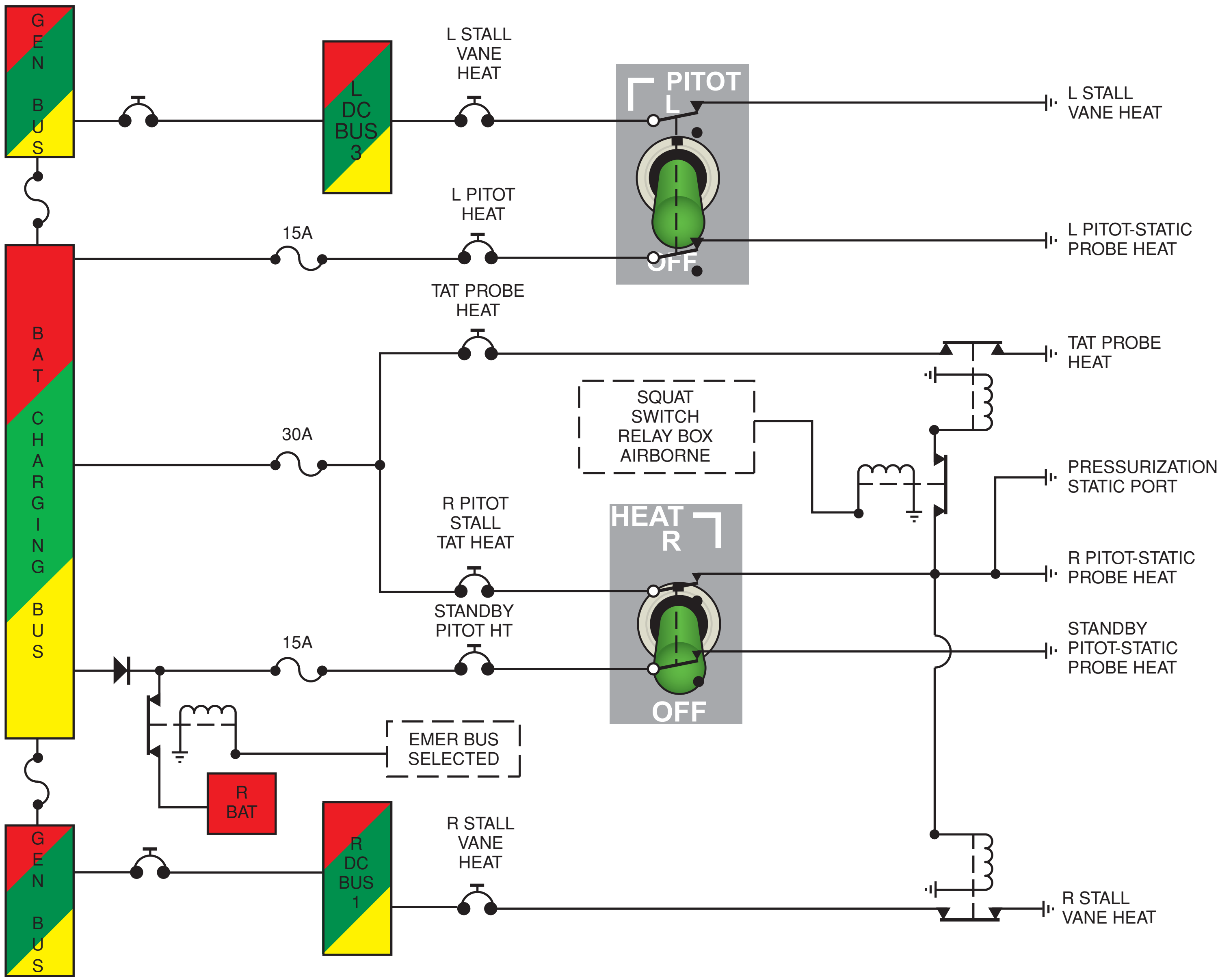

Figure 10-11. Pitot-Static Heat System

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PITOT HEAT Switches

The left and right pitot-static heat, stall warning heat systems, TAT probe heat, and pressurization differential static port heat are controlled with the L and R PITOT HEAT switches located on the ANTI-ICE control panel (see Figure 10-2). The switches have two positions: L-R PITOT HEAT and OFF.

PITOT HEAT Light

The pitot heat monitor system has a single amber PITOT HT light on the glareshield annunciator panel. Illumination of the PITOT HT light indicates that one or more of the pitot-static probe heaters are inoperative or one or both PITOT HEAT switches are off. The pilot would have to cross-check the three pitot-static systems to determine if there was any instrument error due to icing. Some Learjet 60 aircraft are equipped with a three-PITOT HT cluster of lights at the top of the instrument panel. This provides a separate light for each pitot-static probe heat (Figure 10-12).

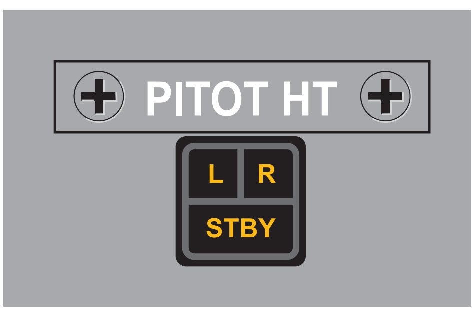

Figure 10-12. Pitot Heat Lights

ROSEMOUNT ICE DETECTOR SYSTEM (OPTIONAL)

The Rosemount ice dector system is installed to detect an icing condition and notifies the pilots by illumination of the amber or white ICE DET lights, in the glareshield panel, and both master CAUT lights. A self-test of the Rosemount ice detector system is conducted every time aircraft power is turned on and the ICE DETECTOR circuit breaker is engaged. The ice detector system self-test will show a failed self-test if the amber ICE DET light and both master CAUT lights are illuminated. The Rosemount ice detection system provides an additional means of ice detection and should not be used as the only source of ice detection. The Rosemount ice detection system receives 28 VDC through the 15-amp ICE DETECTOR circuit breaker on the pilot CB panel.

When the Rosemount ice dector probe detects an icing condition and the STAB WING HEAT switch is OFF, the amber ICE DET light on the glareshield annunciator panel and both master CAUT lights will illuminate (Figure 10-13). Probe deicing is done automatically by the Rosemount system itself. Selecting the STAB WING HEAT switch ON will inhibit the amber ICE DET light and enable the white ICE DET light. The white ICE DET light is an advisory light that illuminates only when icing is detected while the STAB WING HEAT switch is ON. Illumination of the amber ICE DET annunciator with the STAB WING HEAT switch ON indicates a failure of the Rosemount ice detection system.

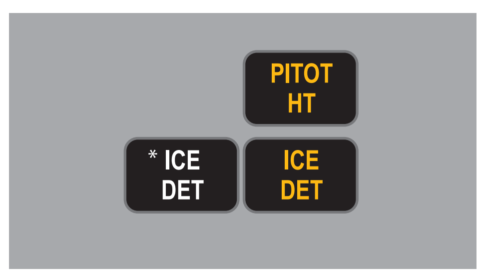

Figure 10-13. ICE DET Annunciator Lights

QUESTIONS

1. Which of the following are heated by engine bleed air?
   A. Pitot-static probes
   B. Nacelle lip heat and engine inner stator vanes and spinner
   C. Inlet pressure and temperature sensors P_1T_1
   D. Stabilizer heat

2. Which of the following is not controlled by the NAC HEAT switches?
   A. Engine inner stator heat
   B. Nacelle heat shutoff valve
   C. Inlet pressure and temperature sensor P.T.
   D. Engine spinner heat

3. The pilot can perform a valid STAB HT system test, in the cockpit, using any of the following sources ***except***:
   A. One or both aircraft generators
   B. GPU
   C. APU
   D. None of the above

4. Which of the following statements is true?
   A. The horizontal stabilizer anti-ice system may be ground checked prior to flight.
   B. When the wing temperature indicator enters the red zone, the amber WING HT light should illuminate.
   C. Icing of wing and/or horizontal stabilizer leading edges affects approach and touchdown speeds and landing distance.
   D. All of the above are correct.

5. With the loss of electrical power, which anti-ice system will still operate?
   A. Nacelle lip heat
   B. Engine inner stator heat
   C. Wing and exterior windshield anti-ice
   D. Both A and B are correct.

6. Which of the following is heated electrically?
   A. Interior windshield defog
   B. Wing leading edges
   C. Exterior windshield anti-ice/defog
   D. Nacelle lip inlets

7. What is the cockpit indication for failure of the nacelle lip anti-ice?
   A. The amber NAC HT light illuminates.
   B. The amber ENG ICE light illuminates.
   C. The green NAC HT ON light extinguishes.
   D. None of the above.

8. Which of the following would cause the PITOT HT annunciator to illuminate?
   A. A pitot heat circuit breaker out and the pitot heat switches are ON.
   B. Circuit breakers in and one or both pitot heat switch(es) OFF.
   C. Circuit breakers in, pitot heat switches ON, and a pitot-static heater element failed.
   D. All of the above are correct.

CHAPTER 11
AIR CONDITIONING

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ILLUSTRATIONS

CHAPTER 11
AIR CONDITIONING

INTRODUCTION

The Learjet 60 air-conditioning system consists of a bleed-air temperature control system, independent distribution systems to the cockpit and cabin for pressurization and ventilation, an auxiliary cabin cooling system (Freon), auxiliary cabin heating and an auxiliary crew heating system.

GENERAL

Primary heating and cooling are accomplished by controlling the temperature of the low pressure manifold bleed air entering the cockpit and cabin distribution systems. The Freon auxiliary cooling system provides additional cooling during both air and ground operations.

The auxiliary cabin heating system provides cabin heating prior to engine start during ground operations.

The ventilating system distributes conditioned bleed air through the cabin and the cockpit. A portion of the bleed air is cooled in an air-to-air heat exchanger. The cooled bleed air is mixed with hot bleed air that bypasses the heat exchanger to obtain the desired temperature for cabin and cockpit distribution. The mixing of the bleed air is regulated by the dual zone (cockpit and cabin) temperature control system. The Freon auxiliary cooling

system and the auxiliary heater are integrated with the ducting used for distribution of conditioned bleed air.

This chapter describes the bleed-air system components and operation (Figure 11-1), starting with the flow control valve and continuing downstream through the heat exchanger to the cabin and cockpit distribution system (Figure 11-2). Both normal and auxiliary heating and cooling systems are described. Refer to Chapter 12—"Pressurization," for information regarding regulation of cabin pressure. Chapter 9—"Pneumatics," describes the engine bleed-air supply and distribution system to the flow control valve.

DISTRIBUTION SYSTEM

This system consists of the flow control valve, the heat exchanger, the cockpit distribution system, and the cabin distribution system.

FLOW CONTROL VALVE

Low pressure manifold bleed air enters the ventilating system through the flow control valve (Figure 11-1). In the event of a cabin pressurization problem, the flow control valve and temperature control ducting in the tailcone can be bypassed and engine bleed air can be admitted directly into the cabin distribution system through the emergency pressurization valves. When engine bleed air enters the cabin through the emergency pressurization valves, it is not cooled and cabin temperature can become very warm.

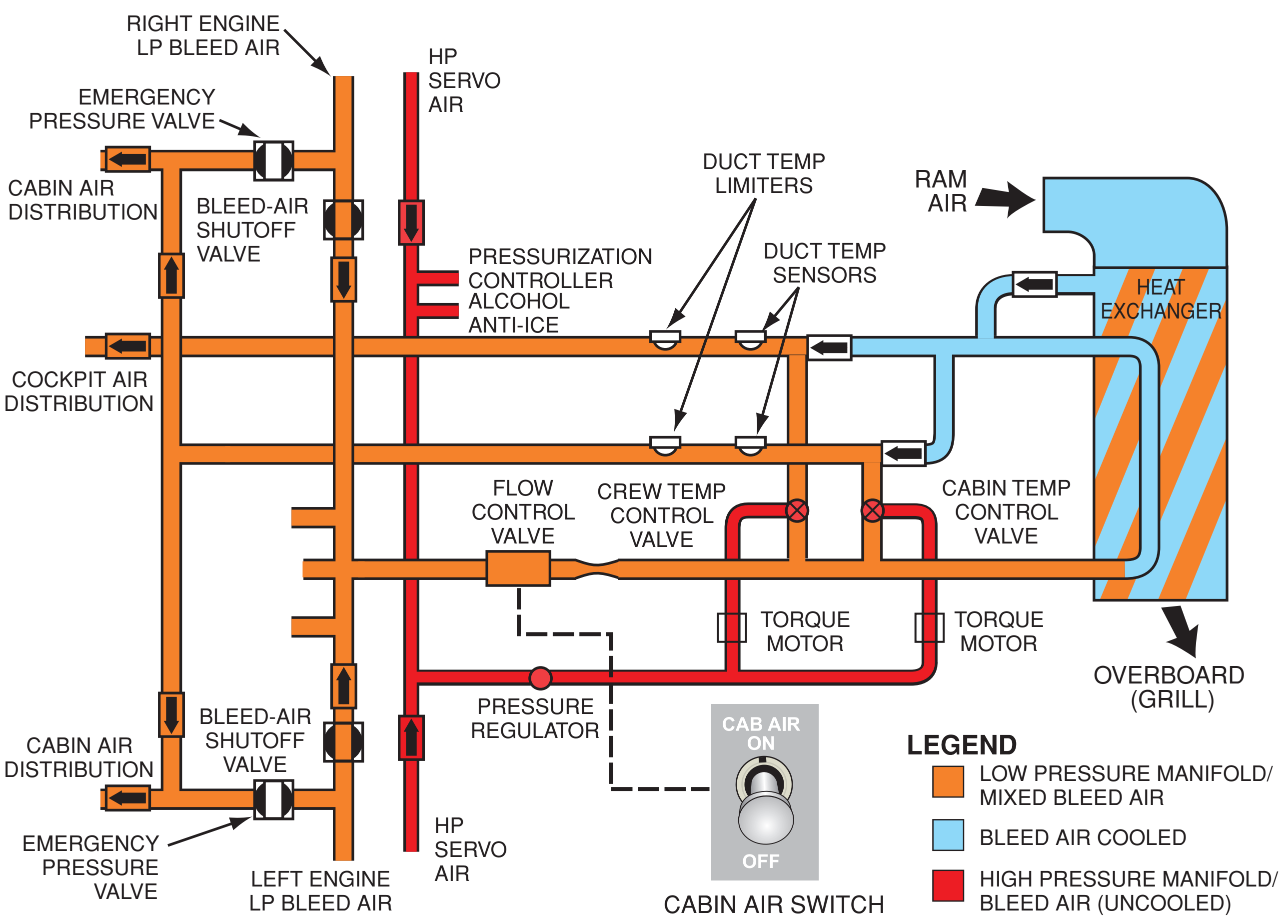

Figure 11-1. Bleed-Air Distribution

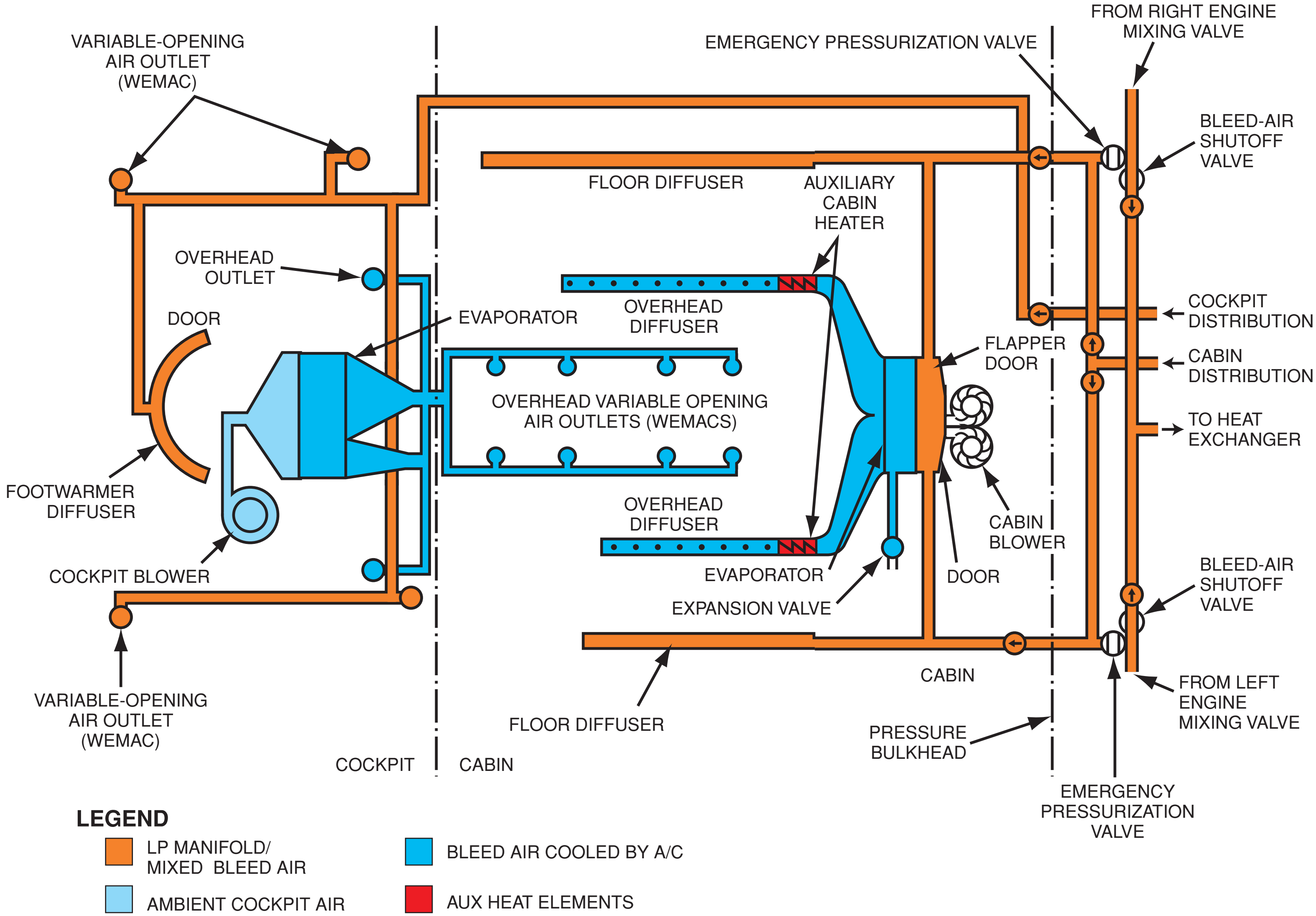

Figure 11-2. Cockpit and Cabin Conditioned Bleed-Air Distribution

The CAB AIR switch on the pressurization control panel controls the flow control valve. With the BLEED-AIR switches on and one or both engines running, setting the CAB AIR switch to ON opens the flow control valve, allowing bleed air to flow to the heat exchanger and the temperature control valves. In the open position, the flow control valve modulates to dampen surges in the bleed-air supply and maintain a constant air flow at all power settings and altitudes.

The flow control valve solenoid requires 28 VDC to close the valve. In the event of electrical failure the flow control valve fails to the open position, regardless of the CAB AIR switch position.

Setting the CAB AIR switch to OFF electrically closes the valve. When the CAB AIR switch is set to OFF, a time-delay circuit prevents the flow control valve from closing for five seconds to minimize pressurization bumps.

HEAT EXCHANGER

Low pressure manifold engine bleed air is ducted through the flow control valve to the air-to-air heat exchanger in the tailcone (see Figure 11-1). When the aircraft is in motion, ambient ram air enters the inlet at the front of the dorsal fin and flows through the heat exchanger where ram air cools engine bleed air. After the air passes through the heat exchanger, it flows overboard through a grilled, circular exhaust port on the bottom center of the tailcone.

CAUTION

> On the ground, do not perform extended engine operation above idle with the CAB AIR and BLEED-AIR switches ON as there is insufficient cooling of the engine bleed air, and possible damage to air-conditioning (Freon) components could result. Tailcone baggage compartment overheating and damage to furnishings and contents could also occur.

When required, ram air can enter the bleed-air distribution ducts through a one-way check valve if no bleed-air pressure is present (i.e., the flow control valve and/or both BLEED-AIR shutoff/regulator valves are closed). The check valve remains closed during pressurization to prevent conditioned air from flowing into the ram air plenum.

Cooled bleed air flows from the heat exchanger in the tail to separate duct systems for distribution to the cabin and cockpit. Regulated amounts of hot bleed air mix with the cooled air from the heat exchanger to control cockpit and cabin temperatures.

COCKPIT DISTRIBUTION SYSTEM

Description

A single duct routes conditioned bleed air from the tailcone to the cockpit, where it is distributed through the outboard variable opening air outlets and the footwarmer diffuser.

Variable opening outlets (WEMACs) are located on each side of the cockpit, aft of the oxygen mask containers, and on each side of the cockpit, near the floor beside the pilot's and copilot's feet (Figure 11-3). The outlets can be

Figure 11-3. Cockpit Distribution System Outlets

adjusted to control the direction and the volume of airflow. The footwarmer diffuser has a series of holes that diffuse conditioned air in the area of crew members' feet.

Cockpit Blower (Crew Fan)

Additional airflow circulation is provided by the cockpit blower located beneath the cockpit floor. The rheostat-type CREW FAN switch (Figure 11-4) on the cabin climate control panel has an OFF detent and is rotated clockwise from the OFF detent to increase the speed of the blower.

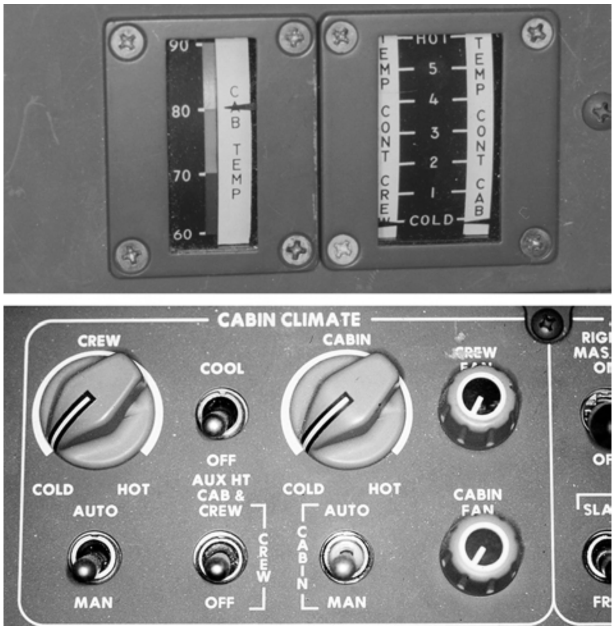

Figure 11-4. Cabin Climate Control Panel

The blower, which operates as part of the Freon cooling system, can be used on the ground or in flight to circulate air. When the cockpit blower is used for cooling, it circulates air from below the cockpit floor to overhead outlets in the cockpit, located above the pilot and the copilot, (Figure 11-3), and to overhead outlets in the cabin (Figure 11-5). All overhead WEMACs (cockpit and cabin) receive their airflow only from the cockpit blower.

An evaporator is encased in the blower ducting. If the Freon system is in operation, air from the cockpit blower is cooled and dehumidified as it flows across the evaporator and then to the overhead WEMACs.

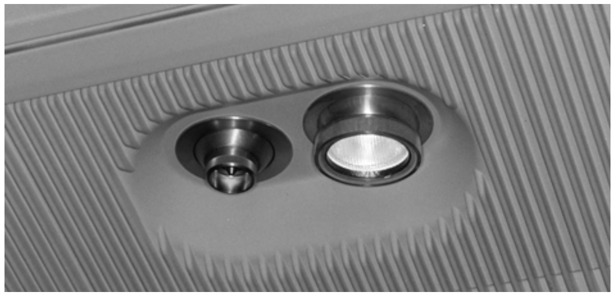

Figure 11-5. Cabin Overhead WEMACs

The cockpit blower can be operated from aircraft battery power to circulate air, but operation of the Freon air-conditioning system on the ground requires a ground power unit (GPU), APU, or at least one operating generator for electrical power to connect the air conditioner.

The cockpit blower is powered by 28 VDC through the cockpit blower control box and the CREW FAN C/B on the copilot ENVIRONMENT circuit-breaker panel. The cockpit blower is deenergized during engine start.

CABIN DISTRIBUTION SYSTEM

Description

Separate ducts from the heat exchanger route conditioned bleed air to the cockpit and cabin.

The distribution duct for the cabin branches into two ducts in the tailcone to deliver conditioned bleed air forward to both sides of the cabin (Figure 11-2). Inside the cabin these ducts branch again to route air down to the cabin floor diffusers and up to cabin overhead diffusers on each side (Figure 11-6). The air routed up to the overhead diffusers passes through a housing which contains an evaporator. If the

Figure 11-6. Cabin Distribution System Outlets

Freon system is operating, the air is cooled enroute to the overhead diffusers.

Air from the cockpit blower is also ducted to the variable opening outlets on the convenience panels above each passenger seat (Figure 11-6).

Cabin Blower (Cabin Fan)

Two inlet ports on the evaporator housing admit air from the cabin blower (cabin fan). A single motor, with a squirrel cage-type blower mounted to each end of the motor, is on the aft side of evaporator housing. Thus, airflow through the housing can come from the bleed-air distribution system when the CAB AIR switch is in the ON position or from the cabin blower.

The cabin fan is off any time the CABIN FAN switch is rotated full counterclockwise or during engine start. Two diverter doors, mounted forward of the cabin fans, are closed whenever the normal bleed-air distribution system is supplying ventilating airflow.

The cabin fan can be operated to circulate air through the overhead diffusers in the cabin. The CABIN FAN variable speed control switch is on the cabin climate control panel beside the temperature control selector knobs (see Figure 11-4).

If electrical power is being supplied by the aircraft batteries only, the cabin blower only operates at a low speed. On aircraft SN 60-045 and subsequent, the cabin blower will not operate on battery power alone. These aircraft require APU, GPU, or generator power to operate the cabin fan.

The cabin blower is automatically energized when the auxiliary heat switch is turned to the CAB CREW position or when the Freon is turned on (COOL position on the COOL–OFF switch) with the CAB AIR switch in the OFF position. The diverter doors forward of the cabin blower are held open by cabin fan airflow.

The cabin blower is powered by 28 VDC through the CABIN FAN circuit breaker on the copilot ENVIRONMENT circuit-breaker panel.

DUAL-ZONE TEMPERATURE CONTROL SYSTEM

DESCRIPTION

The dual-zone temperature control system is designed to allow for independent, automatic control of the temperature of the cockpit and the cabin. The systems can also be operated in the manual mode.

The temperature of conditioned bleed air for cockpit/cabin pressurization and ventilation is controlled by mixing hot bleed air (from the low pressure manifold) with bleed air that has been cooled by passing through the heat exchanger. The hot bleed air bypasses the heat exchanger through two temperature control valves which regulate the amount of hot bleed air entering the cockpit and cabin distribution ducts. The temperature control valves are positioned pneumatically by the electrically-operated temperature control system (Figure 11-7).

Power for the temperature control circuits is 28 VDC supplied through the AUTO TEMP CONT circuit breaker on the copilot ENVIRONMENT circuit-breaker panel group (AUTO mode) and the MANUAL TEMP CONT circuit breaker on the pilot ENVIRONMENT circuit-breaker panel group (MAN mode).

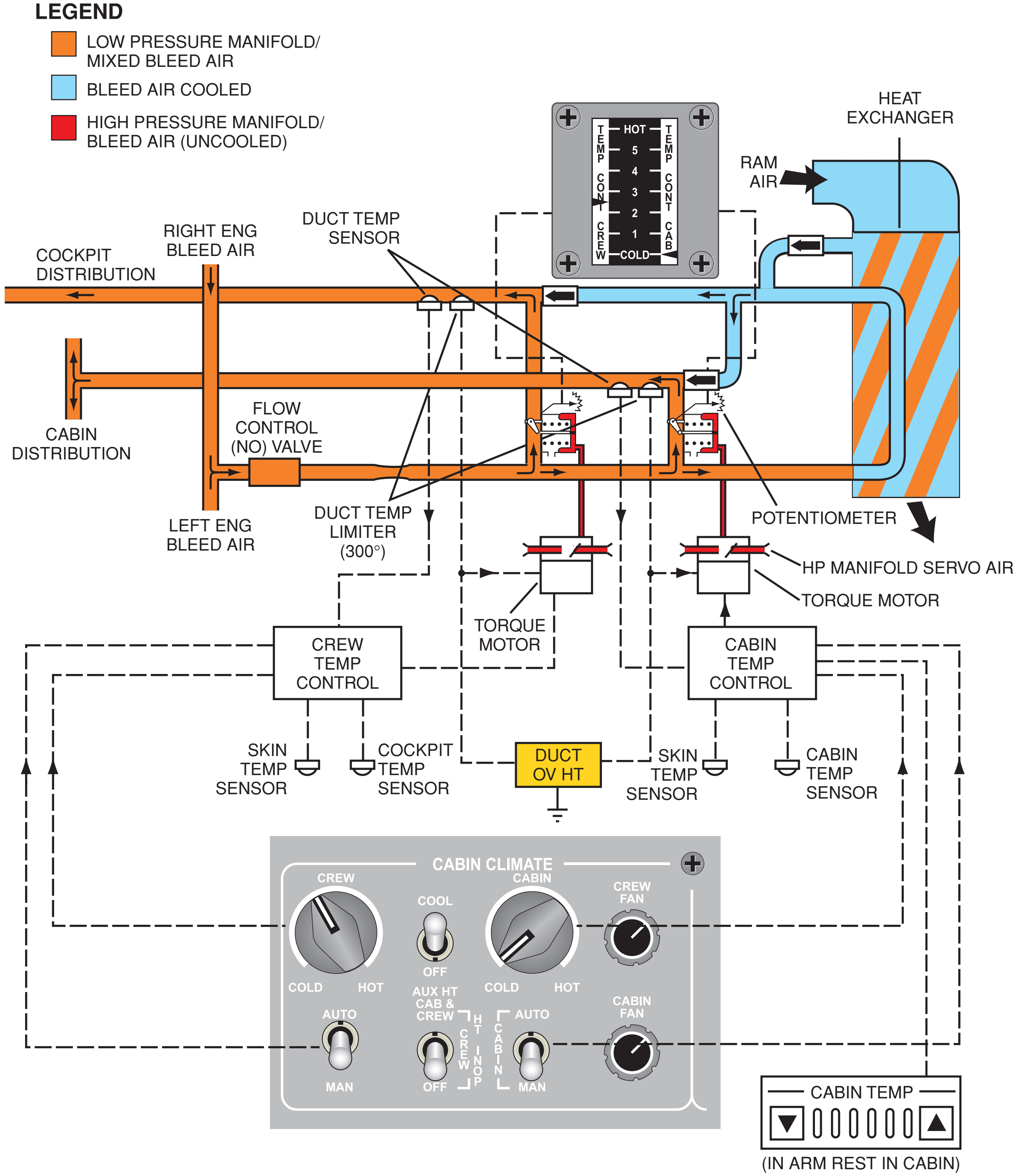

Figure 11-7. Temperature Control System—Controls, Indicators, and Flow Diagram

CONTROLS AND INDICATORS

The controls and indicators for the temperature control systems (Figure 11-7) include the following:

- AUTO–MAN mode switches (two: CREW and CABIN)
- Temperature selectors (two: CREW TEMP and CABIN TEMP)
- Temperature control (valve position) indicators (two: TEMP CONT CREW and TEMP CONT CAB)
- DUCT OV HT annunciator light
- CABIN FAN and CREW FAN switches (one each)
- COOL–OFF switch (Freon control)
- AUX HT CAB–CREW, CREW and OFF (elective Aux HT)

The two AUTO–MAN mode switches are on the CABIN CLIMATE control panel beneath the copilot instrument panel. The cabin temperature control has three positions: AUTO (up), MAN (down), and CABIN (center). Selection of the CABIN position allows temperature control for the cabin to operate in the automatic mode, but with the temperature selected on a remote temperature selector on one passenger armrest in the cabin area instead of the selector in the cockpit.

The CREW and CABIN temperature selectors are rotating knobs with pointers. They are located above the AUTO–MAN switches and turn clockwise from COLD to HOT providing temperature control between 60˚F (cold) and 90˚F (hot).

The CREW and CAB temperature control (valve position) indicators (see Figure 11-7) and cabin temperature indicator (Figure 11-8) are on the center instrument panel above and to the left of the cabin climate control panel. The temperature control indicators are electrically-driven by potentiometers connected to the respective temperature control valves. The indicators operate on 28 VDC supplied through the TEMP CONTROL IND circuit breaker on the pilot ENVIRONMENT circuit-breaker panel. Both temperature control systems include a temperature controller, temperature control valve, a duct temperature sensor, a duct overheat thermostat, and an air and skin temperature sensor for both the cockpit and the cabin areas.

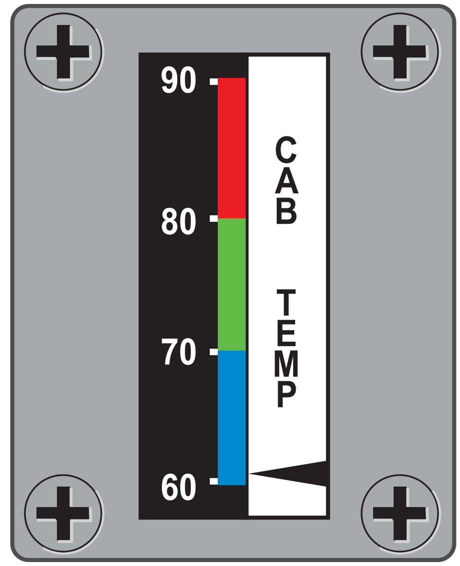

Figure 11-8. Cabin Temperature Indicator

OPERATION

Whenever the cabin or cockpit temperature AUTO–MAN mode switch is set to AUTO, the respective system temperature controller automatically responds to sensor inputs and adjusts the opening of the temperature control valve to maintain the temperature set on the CREW or CABIN temperature selector. Duct temperature limiters are installed in the cockpit and cabin distribution systems to close the affected temperature control valves and light the DUCT OV HT caution light whenever excessively high duct temperatures (300°) are sensed in either system.

Whenever MAN mode is selected with either system AUTO–MAN switch, temperature control valve position is directly controlled by rotating the CREW or CABIN temperature selector. In MAN mode operation, the temperature control valves stay in the position selected on the temperature selector and do not respond to sensor inputs as in the AUTO mode. However, duct overheat protection does function as in the AUTO mode.

COMPONENTS DESCRIPTION

Temperature Controllers

The cockpit and cabin temperature controllers are on the right side of the cabin.

They receive inputs from the AUTO–MAN switches and the temperature selectors. Additionally, they each receive inputs from three temperature sensors: cockpit/cabin temperature sensors, cockpit/ cabin skin temperature sensors, and cockpit/cabin duct temperatures sensors. The sensor inputs are only used during AUTO mode operation.

Temperature Control Valves

The temperature control valves (see Figure 11-7) are operated by regulated pressure from the torque motors. If no pressure is being supplied from the respective torque motor, the valve is held closed by spring pressure.

The temperature controllers regulate the torque motors, which allows HP manifold air to override spring pressure and position the valves to bypass the required amount of hot air (LP manifold) around the heat exchanger.

Loss of power to the torque motors causes flapper valves in the torque motor assemblies to fail closed, shutting off air pressure to the temperature control valves. Without air pressure in the temperature control valves, spring pressure closes the valves (full cold position).

The cabin temperature control valve contains a switch to shut off the Freon system when the cabin temperature control valve is not closed (full cold position).

Duct Temperature Limiter

When operating in AUTO, both crew and cabin temperature controllers respond to signals from their respective duct and skin and air temperature sensors to maintain the selected temperature.

An overheat condition (300°) is prevented by duct temperature limiters (see Figure 11-7) that will deenergize the respective torque motor and illuminate the DUCT OV HT annunciator light. This causes the respective temperature control valve to close. The DUCT OV HT light does not indicate which system, cabin or crew, has malfunctioned, so reference must be made to the temperature control indicators (valve position) to determine the affected system.

When operating in MAN, the temperature controllers adjust the torque motors directly without inputs from their respective temperature sensors. If the CREW or CABIN temperature selector is set too high in manual, the DUCT OV HT annunciator may illuminate. The affected temperature control valve would close as described above in the AUTO mode.

CABIN TEMPERATURE INDICATOR

The cabin temperature indicator (Figure 11-8) is to the left of the temperature control indicators. It provides the crew with a remote indication of cabin temperature. The indicator has three color segments that correspond to a temperature range: blue, 60 to 70˚F; green, 70 to 80˚F; and red, 80 to 90˚F.

The indicator operates on 28 VDC supplied through the CABIN TEMP IND circuit breaker on the copilot ENVIRONMENT circuit-breaker panel.

AUXILIARY COOLING SYSTEM (FREON)

DESCRIPTION AND OPERATION

The refrigeration (air conditioning) system is usually referred to as the Freon system. It is used for ground cooling, in-flight cooling, and cabin dehumidification (Figure 11-9).

There is no altitude restriction on the use of the Freon system, but it is generally used at lower altitudes. Engine bleed-air temperature control is usually adequate at higher altitudes. Using the Freon system during descent decreases humidity and reduces fogging on the inside of the windows.

Freon system operation requires electrical power from a GPU, APU, or an operating engine generator. The control switch for the Freon air conditioner is on the cabin climate control panel. It has two positions: COOL and OFF (Figure 11-9). When the switch is set to COOL, power is supplied to the Freon compressor motor in the tailcone if the following conditions exist:

- APU, GPU, or one generator for ground operation
- Inflight, two generators online and GEN lights out
- STAB WING HEAT switch—OFF
- CABIN TEMP CONT indicator (valve) in COLD (closed) position
- Neither starter engaged
- Freon pressure within limits (hi/low pressure switch)

The cockpit blower beneath the cockpit floor provides airflow across the cockpit evaporator and exhausts it through two overheat outlets above each pilot and through the outlets on the convenience panels above each passenger seat (Figure 11-9). The cockpit blower speed is controlled with the CREW FAN rheostat switch on the cabin climate control panel (Figure 11-9). If the CREW FAN switch is in the OFF detent with the Freon system operating, the cockpit blower will automatically operate at low speed.

Air from the cabin blower (CABIN AIR switch off) or conditioned bleed air (CABIN AIR switch on) flows across the cabin evaporator and exhausts through the overhead diffusers into the cabin.

If the CAB FAN switch is in the OFF position, the cabin blower is automatically energized when the Freon system is turned on if the CABIN AIR switch is off. The speed of the cabin blower can be varied with the CABIN FAN switch on the cabin climate control panel (Figure 11-9).

The Freon system is automatically deenergized during engine start or when the STAB WING HEAT switch is turned on.

Opening the cabin temperature control valve also deenergizes the Freon system.

The compressor motor receives 28 VDC from both generator buses and the battery charging bus through three current limiters and three power relays (Figure 11-10). When the aircraft is on external power, the compressor motor is powered by 28 VDC supplied through a 175-amp current limiter connected to the battery charging bus and a power contactor. When the generators are operating, the compressor motor is powered by 28 VDC supplied through two power contactors and two 175-amp current limiters connected to the generator buses. The power supplied from the generator buses flows through a differential current sensor. If the sensor detects a difference in the electrical load being drawn from the two generator buses, power to all three relays is automatically removed by the control box, and the Freon system shuts off.

In this case, it may be possible to restore air-conditioner operation by cycling the COOL–OFF switch to OFF and back to COOL.

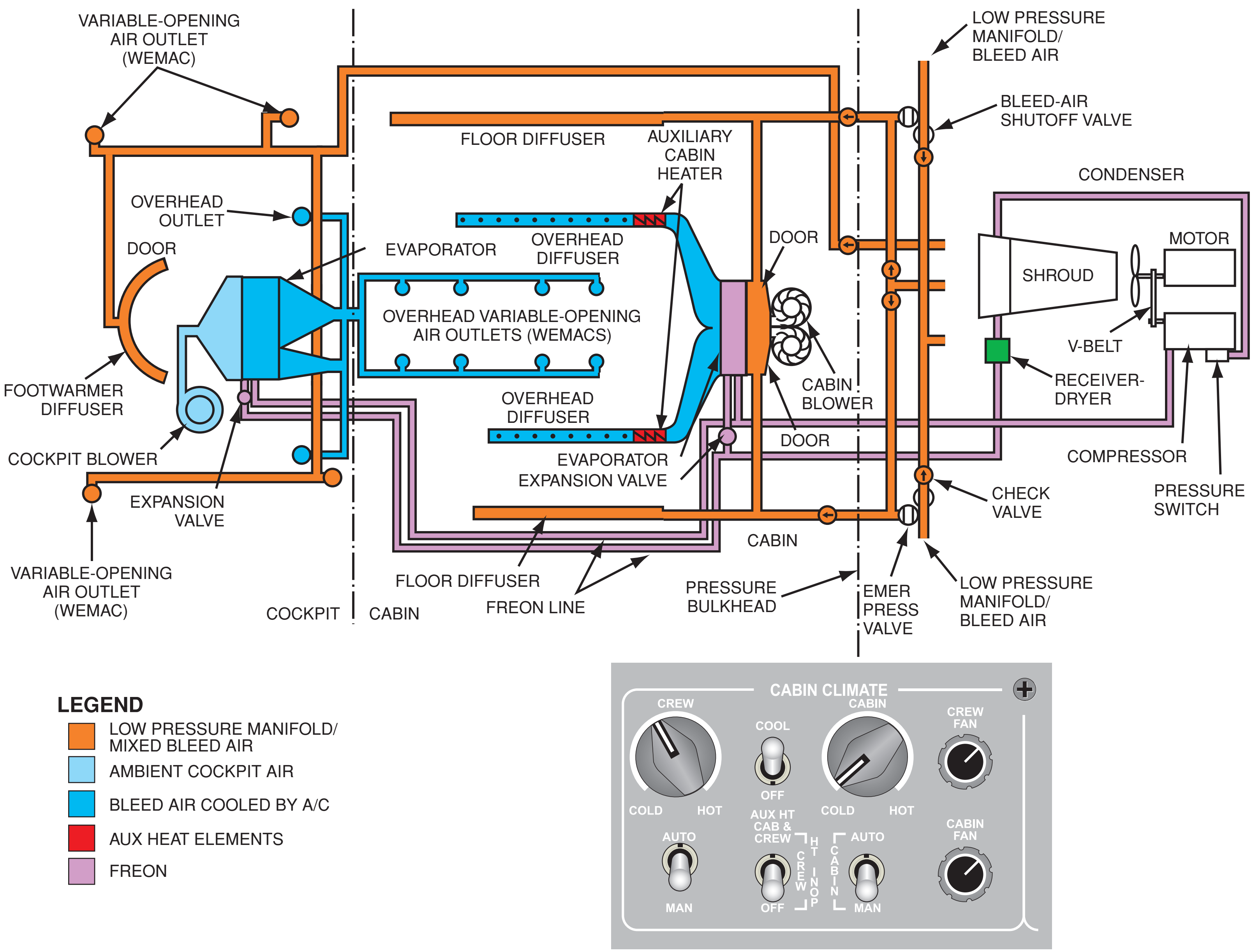

Figure 11-9. Auxiliary Cooling System (Freon) Diagram

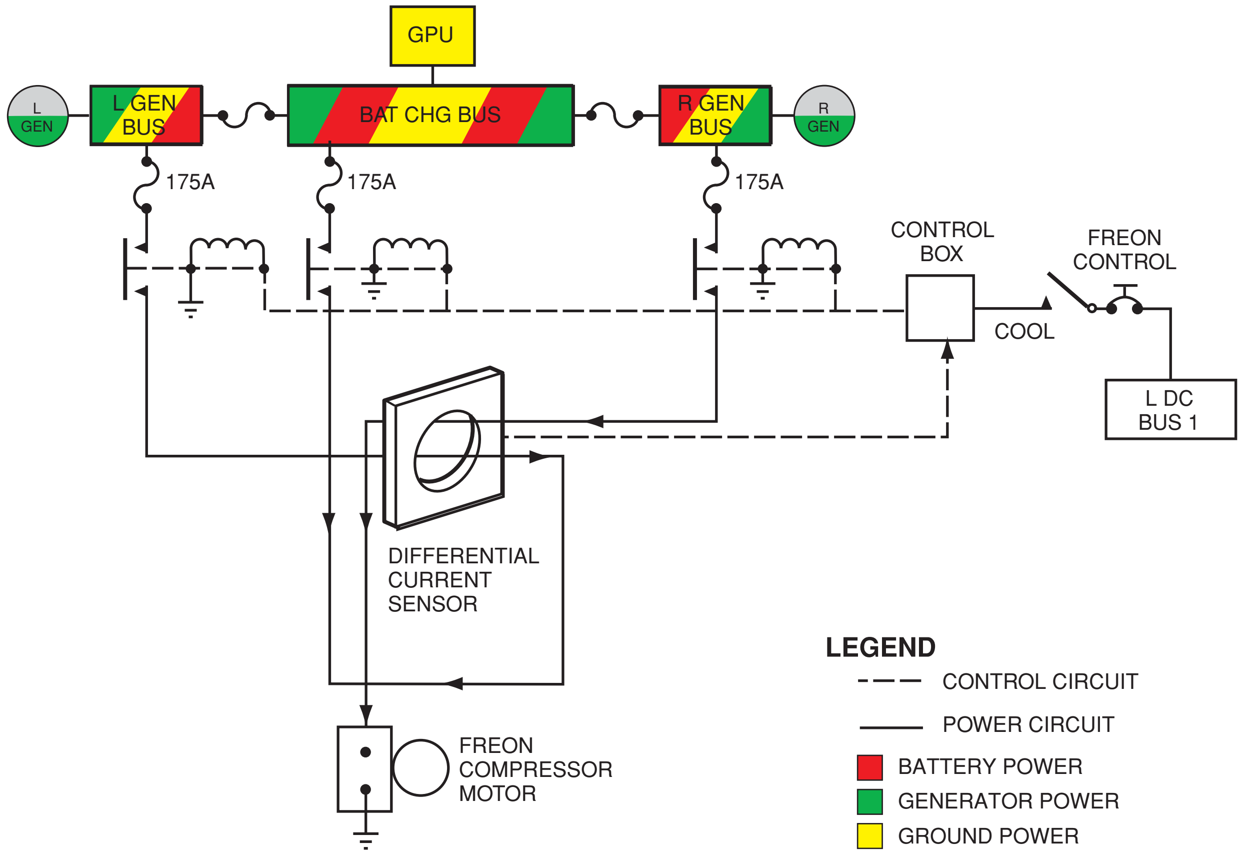

Figure 11-10. Freon Electrical Circuit

NOTE

Caution must be exercised by the crew when operating on GPU or APU power. It takes 28 VDC to close the contacts connecting the Freon air conditioning system. However, should the GPU or APU drop off line, the Freon system will continue to operate off of battery power only. This could cause the batteries to be depleted quickly.

System control circuits are powered by 28 VDC, supplied through the FREON CONTROL circuit breaker on the pilot ENVIRONMENT circuit-breaker panel. The cockpit blower and cabin blower are powered from the battery charging bus through a current limiter and the CREW FAN and CABIN FAN circuit breakers on the copilot circuit-breaker panel.

There are two different Freon air-conditioning systems. Aircraft 60-001 through 60-173 have the R-12 cooling system and operate using the older R-12 Freon. Aircraft 60-174 and subsequent have the R-134A cooling system and operate using the newer R-134 Freon. The R134A cooling system adds two additional WEMACs in the cockpit as pedestal outlets to provide additional cooling air directly to the crew. They are adjacent to the throttle quadrant.

AUXILIARY HEATING SYSTEM

An auxiliary heating system (Figure 11-11) is installed to provide additional cabin heating on the ground. The COOL switch must be in the OFF position in order to operate the auxiliary heater. Power must be supplied by a GPU, APU, or an operating generator.

The system consists of two heater coil assemblies, located in the overhead cabin distribution ducts, two overtemperature thermoswitches, two blower control thermoswitches, two thermal fuses, and an AUX HT switch. The system utilizes the cabin blower to provide heated air circulation through the cabin overhead diffusers.

For ground operation, a GPU, APU, or a generator, is required. The CABIN AUX HT

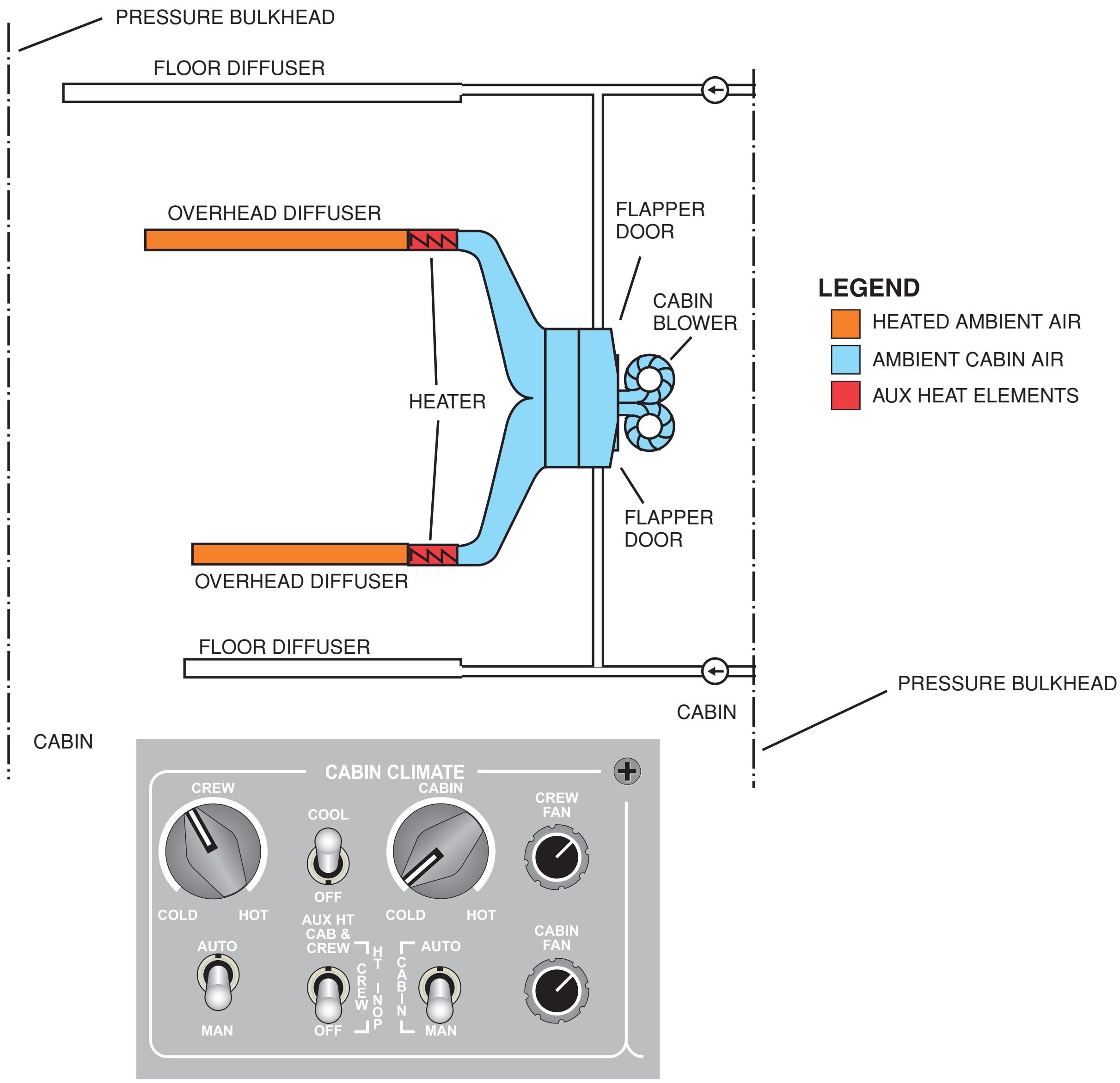

Figure 11-11. Auxiliary Heating Systems (Cabin)

system is rendered inoperative when the CABIN AIR switch is on; therefore, it will not normally be used in flight.

The auxiliary heater is controlled by the three-position AUX HT switch (CAB–CREW/CREW/OFF) on the cabin climate control panel (see Figure 11-11). On aircraft SN 60-067 and subsequent, and on aircraft incorporating SB 60-21-5, selecting the CREW or CAB and CREW position provides direct contact heat for crew foot warming. There are four heaters, one beneath each rudder pedal. Each heater contains two heater blankets and a temperature limiting circuit that controls temperature between 100°F and 130°F independently of the other three heaters. When the temperature of a heater reaches 130°F, a relay removes power to the two heater blankets causing them to cool. On other aircraft, no CREW aux heat is available. In the CAB–CREW position, the cabin blower is energized and operates at the speed selected by the CABIN FAN switch setting. Auxiliary heating is automatically deenergized during engine start or whenever the Freon system is operating. Also, cabin auxiliary heating is deenergized when the CABIN AIR switch is turned on.

Auxiliary heater system control circuits operate on 28 VDC supplied through the AUX CABIN CREW HEAT circuit breaker on the copilot circuit-breaker panel. The heater coils are powered by the battery charging bus through two current limiters. The cabin blower is powered through the CABIN FAN circuit breaker on the copilot circuit-breaker panel.

NOTE

Caution must be exercised by the crew when operating on GPU or APU power. It takes 28 VDC to power up the aux heat system. However, similar to the Freon cooling system, should the GPU or APU drop off line, the aux heat will continue to operate on battery power only.

TAILCONE BAGGAGE COMPARTMENT HEATING SYSTEM

Some Learjet 60 aircraft are equipped with a tailcone baggage heating system that provides for additional heat to warm the compartment if the temperature falls below 35°F. The system is designed to prevent luggage items from freezing and prevent frost from forming on the luggage. Other than turning the system switch on before flight, operation of the system is fully automatic and requires no crew regulation or monitoring.

The system consists of a baggage heater switch, a climate control box, a baggage heater thermostat, and two baggage compartment heaters. The heaters are bonded to Lexan sheet for installation on the floor and sides of the baggage compartment.

The baggage heat switch is in either the tailcone baggage compartment (standard configuration) or on the copilot circuit-breaker panel (optional configuration) (Figure 11-12). The switch controls the 28-VDC power supplied from the battery charging bus to the baggage heaters through a series of relays and switches.

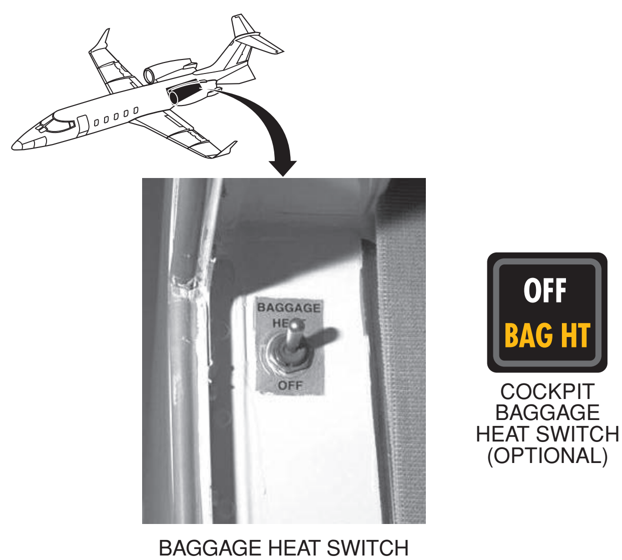

Figure 11-12. Baggage Heat Switch

The baggage heat switch in the tailcone baggage compartment is a two-position (ON–OFF) toggle switch. The optional switch on the copilot circuit breaker panel is a pushbutton switch with lighted captions BAG HT and OFF. Illumination of the BAG HT caption occurs only when the navigation lights switch is set to the ON position. The OFF caption illuminates when the heater switch is in the OFF (not depressed) position and automatically dims when the navigation lights switch is set to the ON position. Both lighted captions illuminate when the WARN LTS press-to-test switch is activated.

There are two baggage heater thermostats in the tailcone baggage compartment that sense the temperature within the compartment when 28-VDC power is supplied to the system. If the temperature in the tailcone falls below 35°F with the baggage heater switch on, the thermostats close and power is supplied to the baggage heater relay to power the heaters. The thermostats open if the temperature rises above 50°F to shut off the heaters.

The tailcone heating system is associated with the auto load-shed relay in the climate control box. If either generator is not online in flight, the tailcone heating system will be inoperative. An engine-driven generator or GPU/APU is required for ground operation. Power is supplied from the battery charging bus through a 50-amp current limiter.

NOTES

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QUESTIONS

1. The crew fan recirculates air to the:
A. Overhead gaspers in the cockpit and WEMACs over each passenger seat in the cabin
B. Side window diffusers in the cockpit
C. Footwarmer diffusers in the cabin
D. Footwarmer diffusers in the cockpit

2. Which of the following conditions will cause the Freon air conditioner to turn off:
A. Single generator operations (in flight)
B. Freon pressure too high/low
C. Cabin temperature control valve not closed
D. All of the above

3. The flow control valve opens when:
A. The BLEED AIR switches are in AUTO
B. CAB AIR switch is ON
C. CKPT AIR switch is ON
D. CAB AIR switch is in AUTO

4. When the aircraft is unpressurized on the ground, air circulation may be provided by:
A. Ram air
B. Crew and cabin fans
C. Bleed-air system
D. Auxiliary defog blower

5. The primary source of cabin heating in flight is:
A. Conditioned engine bleed air
B. Heat pump
C. Auxiliary CREW heat
D. Auxiliary CAB heat

6. Which of the following will prevent the FREON air conditioner from operating?
A. STAB WING HEAT switch is in the STAB WING HEAT position.
B. Cockpit temperature control valve is above COLD.
C. Air cycle machine
D. Auxiliary CAB heat is ON.

7. Illumination of the DUCT OV HT light indicates:
A. Overtemp in the engine pylon
B. Overtemp in the cabin diffusers
C. Overtemp in the auxiliary CABIN heat
D. Overtemp in the cockpit or cabin distribution ducts in the tailcone

8. With the Freon air-conditioning system in operation, the CREW FAN operates at a minimum speed if the CREW FAN switch is in the _______ position.
A. OFF (full counterclockwise)
B. Mid-range
C. High speed (full clockwise)
D. AUTO

9. In order to operate the AUX HT CAB heater:
A. GPU/APU or generator must be on.
B. COOL-OFF switch (Freon) must be OFF.
C. CABIN AIR switch must be OFF
D. All the above

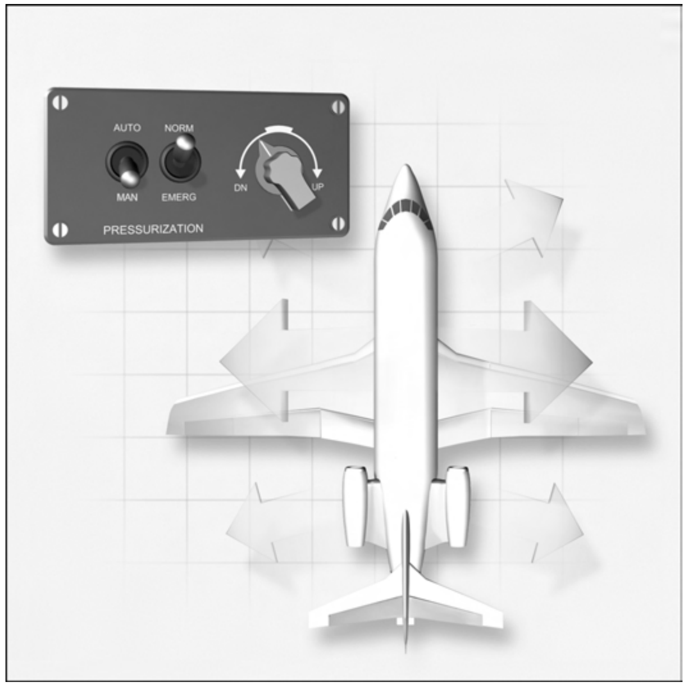
CHAPTER 12
PRESSURIZATION

CONTENTS

ILLUSTRATIONS

TABLES

CHAPTER 12 PRESSURIZATION



INTRODUCTION

The Learjet 60 pressurization system maintains a cabin altitude lower than actual aircraft altitude. This is accomplished by conditioned bleed air entering the cabin and cockpit areas from the flow control valve and temperature control system or emergency pressurization valve(s), through the airflow outlets, and controlling the amount of air exhausted overboard.

GENERAL

The pressurized area extends from the forward pressure bulkhead frame 6, to the aft pressure bulkhead frame 26A. The inflow of air from the flow control valve is consistent through a wide range of power settings, and the outflow is controlled by two outflow valves on the forward pressure bulkhead. Positioning of the outflow valves is controlled by an electronic cabin pressurization controller or a backup manual controller, both in the cockpit. During emergency pressurization, the air entering the cabin and cockpit bypasses the flow control valve and temperature control systems.

MAJOR COMPONENTS

The pressurization control system incorporates the following major components:

- Cabin pressure selector panel
- Cabin pressure controller
- Cabin pressure indicator
- Primary outflow valve
- Secondary outflow valve
- Vacuum jet pump
- Vacuum regulator

PRESSURIZATION CONTROL PANEL

The pressurization control panel provides crew-to-system interface capabilities for all modes of system operation.

CABIN AIR SWITCH

The CABIN AIR switch (Figure 12-1), located on the left side of the pressurization control panel, removes DC power from the flow control valve when selected ON. This will open the flow control valve allowing conditioned bleed air to enter the cabin.

LANDING ALTITUDE SELECTOR

The landing altitude selector knob, located on the pressurization control panel (Figure 12-1), is used to position the pointer on the LDG ALT scale to the destination field elevation. The selector knob has the letter A on it and is located just below and left of the circular LDG ALT scale. Destination field elevation is normally selected with this control before takeoff. Set field elevation, not field pressure altitude. The controller automatically compensates for nonstandard field pressure through the ADC input. The ADC receives this information when the crew enters the altimeter setting on the primary flight display (PFD).

MODE SWITCH AND MANUAL MODE CONTROLS

A latching-type mode switch is located on the pressurization control panel (Figure 12-1) that allows the operator to select the manual mode of operation if desired. The bottom half of the switch has a white MANUAL annunciator which illuminates when manual is selected by depressing the mode switch. If there is no fault present in the controller, the operator can return to the automatic mode by depressing the mode switch a second time. The upper half of the mode switch contains an amber FAULT light that will illuminate if the automatic system faults while in the automatic mode of operation.

The manual controller (MAN ALT) is a three-position lever to the left of the MODE switch. When in the manual mode, the MAN ALT lever may be placed in the UP or DN position (as desired). Then, the cabin altitude rate of change (vertical speed) may be controlled with the MAN RATE knob just below it. When the cabin reaches the desired altitude, the MAN ALT lever should be placed back to the center position.

The cabin controller can be operated in the manual mode with or without aircraft electrical power; however, the cabin pressure indicator is inoperative with loss of electrical power.

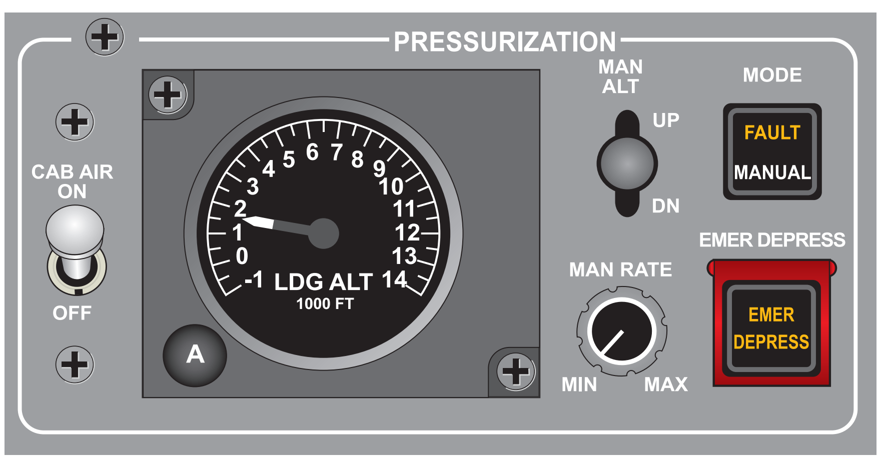


Figure 12-1. Pressurization Control Panel

The electronic controller features built-in test equipment which performs fault detection and annunciation routines during ground and flight operations. Should a fault be detected, the FAULT light illuminates and the system reverts to the manual mode if any of the following occur: (1) the cabin pressure controller detects a fault, (2) electrical power is lost to the cabin pressure controller, or (3) the cabin altitude has reached 8,600 ±250 feet (LDG ALT set to 8,000 feet or less). In addition to the FAULT light, the amber PRESS SYS light on the annunciator panel will illuminate. When the mode switch is depressed, the FAULT and PRESS SYS lights will extinguish and the MANUAL annunciator in the mode switch will illuminate. This is an indication and a reminder to the crew that the manual mode of operation has been selected.

Tables 12-1 and 12-2 show a summary of events that occur if the cabin climbs to abnormally high altitudes.

Table 12-1. PROTECTION AND WARNING FEATURES (NORMAL)

LDG ALT SET AT 8,000 FEET OR LESS	
CABIN ALTITUDE	**PROTECTION AND WARNING**
14,500 ±250	• Passenger oxygen masks deploy. • Cabin overhead panel lights illuminate (see Chapter 17, Miscellaneous Systems).
13,700 ±500	• Cabin altitude limiters close outflow valves.
10,100 ±250	• Cabin altitude warning horn sounds. —Initiate emergency descent. —Mute horn with mute switch. • On aircraft 60-271 and subsequent and prior aircraft modified by SB 60-31-1, a CABIN ALT HI light will illuminate.
9,500 ±250	• Emergency pressurization (automatic) is activated, directing bleed air directly into the cabin. • EMER PRESS annunciator illuminates (see Chapter 9, Pneumatics).
8,750 ±250	• If MANUAL has been previously selected on the pressurization control panel, the PRESS SYS annunciator illuminates.
8,600 ±250	• Pressurization control automatically reverts to manual mode. • PRESS SYS annunciator illuminates. • FAULT annunciator illuminates.

NOTE:
All the pressurization, protection, and warning features require electrical power for operation except the cabin altitude limiters. In the event of electrical failure, the automatic mode and emergency pressurization mode are not available. Manual control is still available, but the cabin altitude indicator will be inoperative (no 10,100 ft high altitude horn or automatic mask/light deployment). Passenger oxygen masks can be deployed manually.

Table 12-2. PROTECTION AND WARNING FEATURES (HIGH ALTITUDE AIRPORT)

LDG ALT ABOVE 8,000 FEET	
Prerequisites: (1) In automatic mode on controller (2) Destination LDG ALT set above 8,000 feet (3) Aircraft has descended 1,000 feet (4) Aircraft is below 25,000 feet	
CABIN ALTITUDE	**PROTECTION AND WARNING**
14,500 ±250	• Passenger oxygen masks deploy. • Cabin overhead panel lights illuminate (see Chapter 17, Miscellaneous Systems). • Cabin altitude warning horn sounds (CABIN ALT HI light illuminates). • Emergency pressurization activates. • EMER PRESS annunciator illuminates (see Chapter 9 Pneumatics). • Fault annunciator illuminates. • Pressurization control reverts to manual mode. • PRESS SYS annunciator illuminates.
13,700 ±500	• Cabin altitude limiters close outflow valves.

EMER DEPRESS SWITCH

The EMER DEPRESS switch, located below the mode switch on the pressurization control panel, is guarded with a hinged, red-bordered, transparent cover. Lifting the cover and depressing the switch will cause the aircraft cabin to rapidly depressurize to a maximum altitude of 13,700 feet (automatic outflow valve limiters activate).

When the switch is depressed, an amber EMER DEPRESS annunciator in the switch illuminates, and the cabin pressure controller is signaled to drive both outflow valves to the full open position. The cabin will climb at a rapid rate until the aircraft is unpressurized or the cabin reaches 13,700 feet, whichever comes first. If the cabin altitude reaches 13,700 feet, altitude limiters on each outflow valve will cause the outflow valves to move toward closed to keep the cabin from climbing higher. Depressing the EMER DEPRESS switch a second time will return the system to the previous mode, and the EMER DEPRESS annunciator will extinguish. EMER DEPRESS will work in either the manual or automatic mode. However, it does not work if electrical power is not available to the cabin pressurization controller.

CABIN PRESSURE CONTROLLER

The cabin pressure controller (Figure 12-2) is a microprocessor-based electronic unit. It is located behind the copilot instrument panel and is electrically connected through wiring to the pressurization control panel. It is used to control cabin pressure when the system is in the automatic mode of operation.

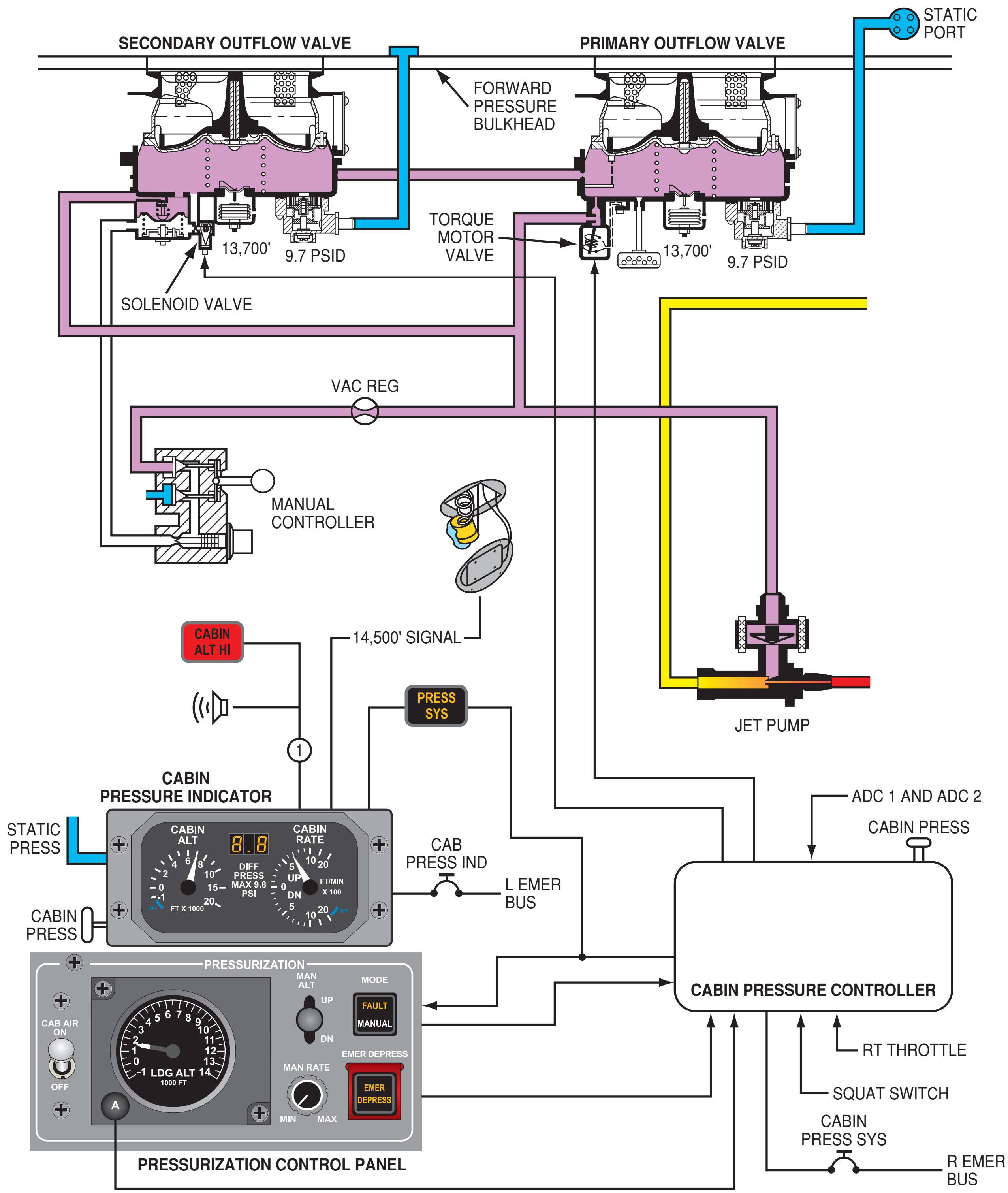


Figure 12-2. Pressurization Control System Schematic

Table 12-3. CONTROLLER CLIMB/DESCENT SCHEDULE

AIRCRAFT ALTITUDE, FEET	CLIMB SCHEDULE, FEET	DESCENT SCHEDULE, FEET
51,000	8,000	8,000
49,000	7,767	7,645
47,000	7,414	7,225
45,000	7,061	6,770
43,000	6,708	6,280
41,000	6,355	5,740
39,000	6,002	5,100
37,000	5,649	4,620
35,000	5,296	4,140
33,000	4,943	3,680
31,000	4,590	3,210
29,000	4,237	2,660
27,000	3,883	2,110
25,000	3,530	1,340
23,000	3,177	620
22,000	3,001	–170
21,000	2,824	–970
19,000	2,471	–1,600
17,000	2,118	–1,600
15,000	1,765	–1,600
13,000	1,412	–1,600
11,000	1,059	–1,600
9,000	706	–1,600
7,000	353	–1,600
5,000	–120	–1,600
3,000	–620	–1,600
1,000	–1,110	–1,600
0	–1,360	–1,600
–1,000	–1,600	–1,600

When DC electrical power is applied to the aircraft, the cabin pressure controller performs a functional check of the system. Once the functional check is successfully completed, the system is in the automatic mode, and the only input required by the crew will be to set landing field elevation.

Should the system fail the functional self-test, or fault while in the automatic mode, the FAULT light, on the pressurization control panel, and the PRESS SYS light, on the annunciator panel, would then illuminate and the system would revert to the manual mode.

The operator should then depress the mode switch, which will cause the FAULT and PRESS SYS lights to go out, and the MANUAL light to illuminate.

The controller utilizes inputs from the pressurization control panel, along with aircraft altitude and barometric correction from the air data computers (ADCs). Other inputs received are: cabin pressure, thrust lever position (82% N_1) from the right thrust lever, and landing gear squat switches.

The controller uses the inputs and compares them to the internal program logic. These inputs are constants, and are continuously updated in the controller memory, which makes the system fully automatic.

In flight, in the automatic mode, the controller uses either the climb or descent schedule (Table 12-3) to maintain the appropriate cabin altitude. Rate-of-climb or descent in the cabin is nonselectable in the automatic mode and is determined by the controller from a maximum of 600 fpm climb to 375 fpm descent rate.

PRIMARY OUTFLOW VALVE

The primary outflow valve (see Figure 12-2) is a spring-loaded closed, pneumatic poppet-type valve. The valve opening is controlled by regulating the vacuum pressure in the control chamber of the valve. The vacuum (low pressure) is regulated through an electronic torque motor valve, which receives signals from the cabin pressure controller. The valve has a maximum differential pressure relief valve, set at 9.7 psid, that overrides the pneumatic control vacuum to limit the cabin-to-ambient differential pressure below the structural limits of the fuselage.

Also included, is an altitude limiter set at 13,700 feet cabin altitude and a proximity switch that signals the controller when the outflow valve is full open.

SECONDARY OUTFLOW VALVE

The secondary outflow valve is also a spring-loaded closed, pneumatic poppet-type valve that is very similar to the primary outflow valve.

In the automatic mode of operation, the secondary outflow valve is slaved to the primary outflow valve so that both valves will operate together. This is accomplished by interconnecting the control chambers of the two valves with an unrestricted, open tube. The secondary outflow valve contains a solenoid-controlled valve which is energized open while in the automatic mode of operation. This disables the manual mode controls. When the controller faults to manual, or MANUAL is selected, the solenoid-controlled valve is deenergized closed. This allows the secondary outflow valve to be positioned by the manual controller and the primary outflow valve is now slaved to the secondary outflow valve.

VACUUM JET PUMP

The vacuum jet pump (see Figure 12-2) is located in the tail section of the aircraft and is driven by HP bleed air from either engine. It provides a source of vacuum (low pressure) to the torque motor valve on the primary outflow valve for automatic operation and to the secondary outflow valve for manual operation. Vacuum is also directed through a vacuum regulator to the UP port of the manual controller.

VACUUM REGULATOR

The vacuum regulator (see Figure 12-2) is designed to maintain a constant vacuum source of approximately .5 psig below cabin pressure. This regulated vacuum is used by the manual controller to control the cabin during manual operation.

PRESSURIZATION CONTROLLER POWER SOURCE

The cabin pressurization controller is powered from the right EMER BUS through the CABIN PRESS SYS circuit breaker located in the ENVIRONMENT group of circuit breakers on the copilot circuit-breaker panel. If the circuit breaker pops or electrical power is lost to the right EMER BUS, the pressurization controller reverts to the manual mode. The MAN annunciator on the pressurization control panel may or may not be illuminated depending on the extent of electrical power failure. Also, the cabin pressure indicator may or may not be working depending on the extent of electrical power failure.

CABIN PRESSURE INDICATOR

The cabin pressure indicator (Figure 12-3), located above the pressurization control panel, contains the cabin altimeter, cabin vertical speed indicator, and a LED readout of cabin differential pressure. The cabin pressure controller is independent of the cabin pressure indicator and will continue to function in the automatic mode with failure of the cabin pressure indicator.

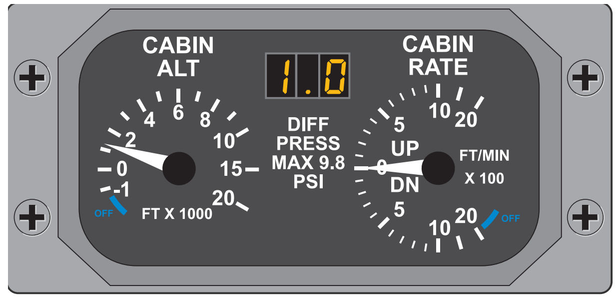


Figure 12-3. Cabin Pressure Indicator

The cabin pressure indicator module senses cabin pressure through a filtered port on the indicator case and displays it on the CABIN ALT scale. The indicator displays cabin altitude rate of change on the CABIN RATE indicator. The cabin pressure indicator module senses atmospheric pressure through a connection to the static pressure line that serves the ADC. Based on the existing cabin pressure and atmospheric pressure, the indicator module calculates the differential pressure and displays it in the digital DIFF PRESS window.

Due to the limiting parameters set in the controller software program and a 9.7 psid pressure differential relief valve on each outflow valve, the differential pressure should never exceed 9.7 psid; however, if the differential pressure does exceed 9.8 psid, the digital readout will flash and the cabin pressure indicator module will illuminate the amber PRESS SYS annunciator to alert the crew (see Figure 12-2). The cabin pressure indicator will also cause the PRESS SYS light to illuminate if the pressure differential exceeds a negative 0.5 psid.

In the unlikely event that the differential pressure exceeds 9.9, the first digit of the display is omitted. For example, a 10.1 pressure differential would be displayed as 0.1 psid. Since differential pressure indication above 9.8 psid flash, it would be easy to differentiate between 10.1 psid and 0.1 psid even though both would be displayed as 0.1 psid.

In addition to the indications on the cabin pressure indicator described above, the module also generates a signal at 8,750 ±250 feet that will cause the PRESS SYS light to illuminate if in the manual mode of cabin pressure control. An additional signal is generated at 10,100 ±250 feet that activates the cabin altitude warning horn (auto or manual mode), and a final signal at 14,500 ±250 feet, which causes the passenger oxygen masks to drop and turns on the cabin overhead lights (auto or manual mode). The signal to activate the cabin altitude warning horn is reindexed to 14,500 feet under the following conditions: the LDG ALT is set above 8,000 feet, the aircraft has descended 1,000 feet, and the aircraft has descended below 25,000 feet. This is to avoid an unnecessary warning horn if landing at a high field elevation.

CABIN ALTITUDE WARNING HORN

The cabin altitude warning horn will activate if the cabin altitude reaches 10,100 feet, or 14,500 feet, depending on the conditions described above. See Tables 12-1 and 12-2 for a summary of other events that will occur if the cabin reaches abnormally high altitudes.

The horn can be silenced by depressing the mute switch in the knob of the right thrust lever. When muted, it will remain silenced for 60 seconds, and then will reactivate if the cabin is still above the altitude at which it was triggered. The CABIN ALT position on the rotary test switch is used to test the cabin altitude warning horn (and mute function).

CABIN ALT HI LIGHT

On aircraft 60-271 and subsequent and prior aircraft modified by SB 60-31-1, a red CABIN ALT HI light will illuminate in conjunction with the cabin altitude warning horn when the cabin altitude reaches 10,100 ±250 ft.

CABIN PRESSURE INDICATOR POWER SOURCE

The cabin pressure indicator is electrical and receives power from the left emergency bus through CAB PRESS IND circuit breaker, in the ENVIRONMENTAL group of circuit breakers, on the pilot side. If electrical power is lost to the indicator, the indications and functions of the cabin pressure indicator will be lost. Lost functions include: (1) high altitude warning horn/light, (2) automatic deployment of passenger masks/overhead lights.

SYSTEM OPERATION

AUTOMATIC MODE

The cabin pressure controller completes a self-test when DC electrical power is applied to the aircraft electrical system. A successful completion of the test is indicated by no FAULT light in the control panel MODE switch. If the FAULT light does illuminate, a second self-test can be initiated by pulling and resetting the cabin PRESS SYS circuit breaker in the ENVIRONMENTAL group of circuit breakers on the copilot circuit-breaker panel.

While accomplishing the preflight checks, the crew will normally enter the planned destination field elevation (LDG ALT) on the pressurization control panel, check to ensure that the MAN ALT lever is in the center position, the MAN RATE knob is in the MIN position, and that MANUAL is not selected on the MODE switch.

When the cabin air switch is turned on before takeoff and the thrust levers are still below 82% N_1, the cabin pressure controller is in the ground mode (Figure 12-4). The controller will drive the outflow valves full open, but the cabin will rate down approximately 80 to 100 feet.

When the right thrust lever is advanced past approximately 82% N_1 for takeoff, the system enters a prepressurization mode (Figure 12-5). The controller starts rating the cabin down an additional 150 feet, not to exceed 1,000 fpm. This prepressurization eliminates any pressure transients during rotation by allowing the outflow valves to attain a controlling position before lift-off. If the thrust levers are retarded below 82% N_1, and the aircraft is still on the ground, as in an aborted takeoff, the cabin controller will rate the cabin back up at 500 fpm to the ground mode level (Figure 12-6).

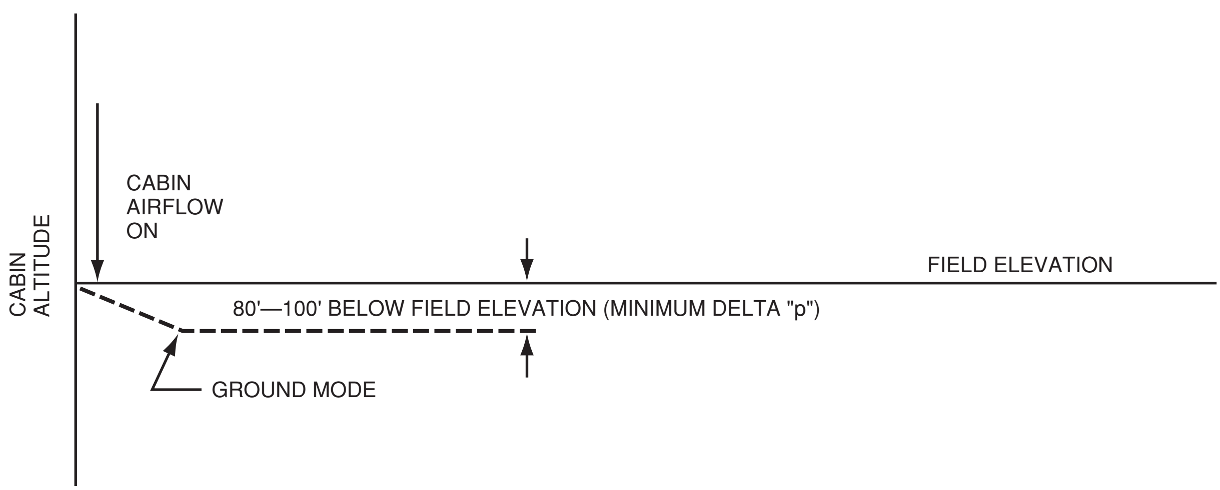


Figure 12-4. Controller—Ground Mode

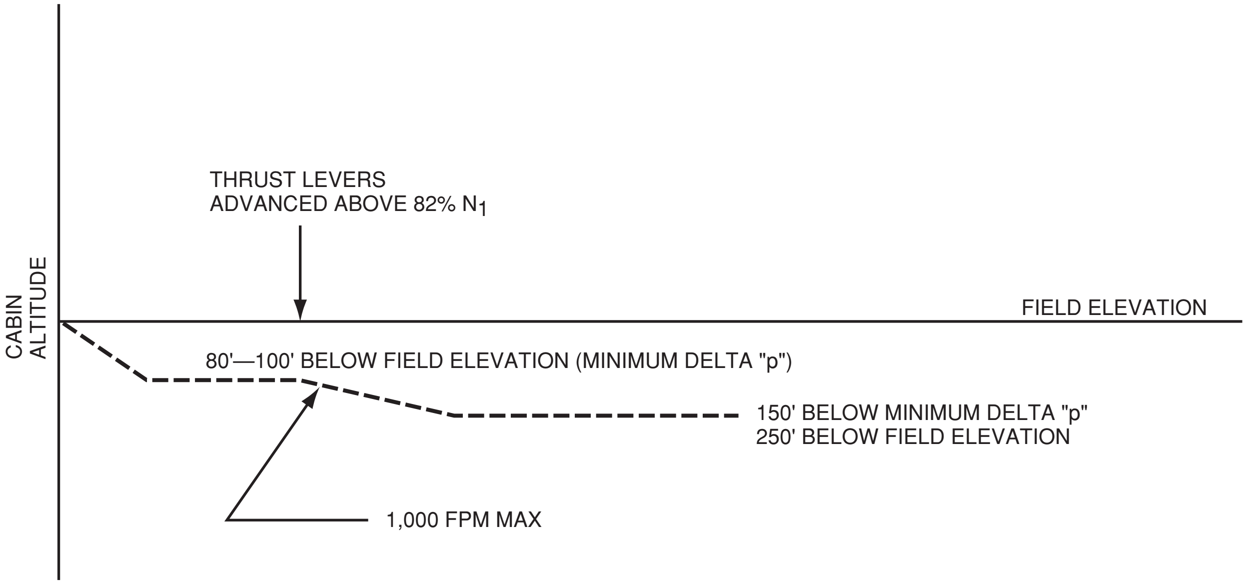


Figure 12-5. Controller—Prepressurization

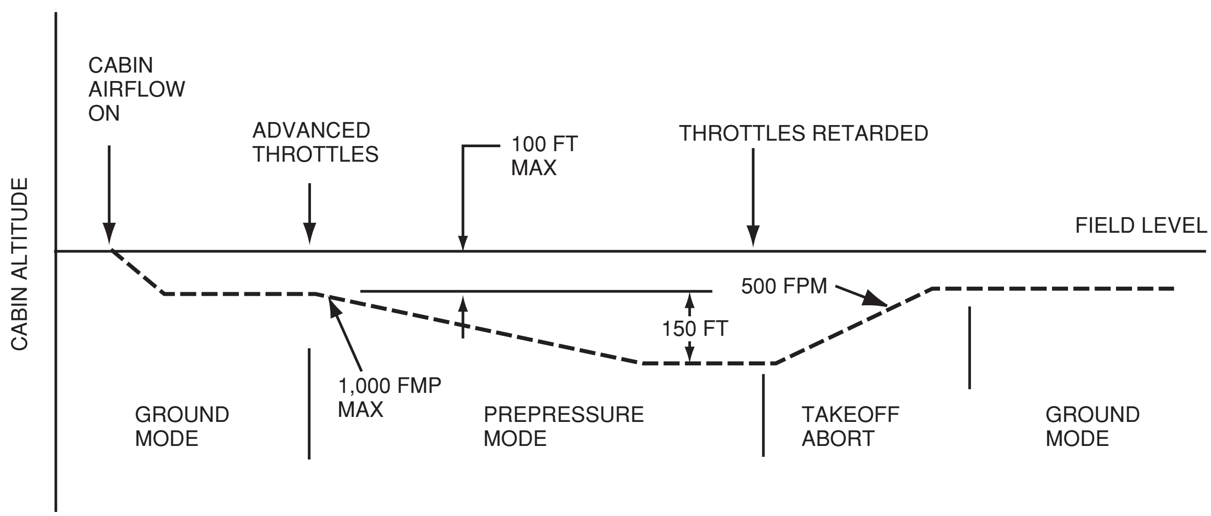


Figure 12-6. Controller—Takeoff Abort

When the aircraft becomes airborne, the system is in the flight mode. The cabin altitude will remain at the existing cabin altitude at lift-off until the aircraft intercepts the climb schedule (Figure 12-7).

During the climb mode operation, a fixed schedule of cabin altitude versus aircraft altitude is used for automatic control of cabin pressure. This fixed schedule is programmed into the controller and is referred to as the climb schedule (see Table 12-3). The climb rate is accomplished through a continuous update of true static pressure from the ADCs to the controller at a maximum rate of 600 fpm.

If taking off from a sea level airport, for example (Figure 12-7), the cabin would be at approximately minus (–) 250 feet at lift-off. Referring to Table 12-3, you will note that the aircraft altitude is approximately 4,500 feet at a point adjacent to –250 feet in the climb schedule column (requires interpolation). Therefore, as the aircraft climbs through 4,500 feet, the cabin will begin to climb and maintain the climb schedule in Table 12-3. If the aircraft is leveled-off at an intermediate altitude of FL 210, for example, the cabin will level at 2,824 feet. If you are later cleared to climb to FL 410, the controller will resume the climb schedule as soon as the aircraft has climbed 200 feet, and when the aircraft is leveled-off at FL 410, the cabin should level at 6,355 feet. The cabin controller will maintain the climb schedule until descent is begun.

If an inflight abort is necessary after takeoff, the controller will retain the takeoff field altitude in a stored memory. This feature allows the cabin pressure to return to the takeoff field altitude if the aircraft descends 1,000 feet within 10 minutes after takeoff and did not climb more than 6,000 feet above the takeoff field elevation before descending (Figure 12-8).

When descent is initiated from normal cruise altitude and the aircraft descends 1,000 feet, the controller will establish the cabin altitude on the descent schedule (see Table 12-3) and maintain that schedule. However, the cabin descent rate should not exceed 375 fpm. At some point, the cabin altitude will approach the selected landing field elevation and maintain this altitude during the approach to landing (Figure 12-9).

If the LDG ALT, set in the controller, is higher than the cabin altitude, the controller will rate-up upon initiation of the descent, at a maximum rate of 150 fpm.

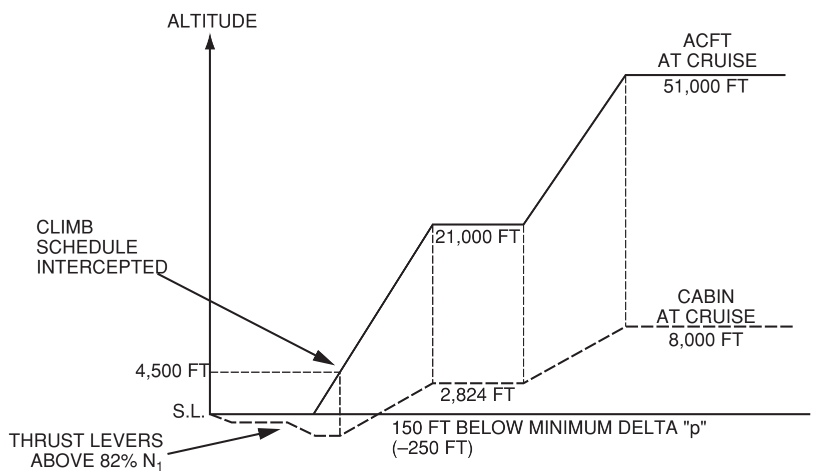


Figure 12-7. Controller—Takeoff and Climb

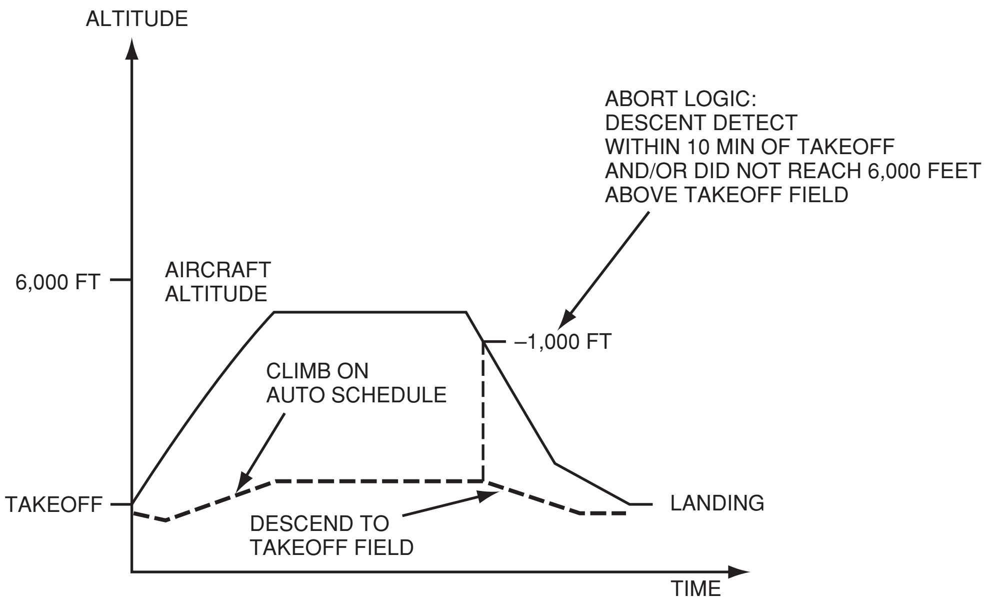


Figure 12-8. Controller—Flight Abort Mode

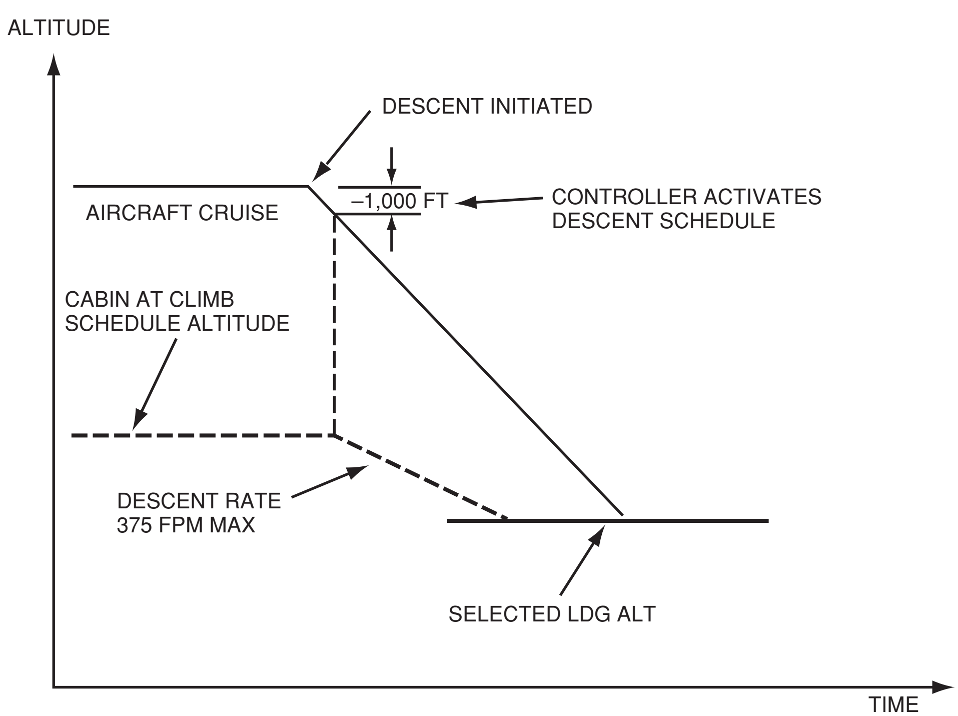


Figure 12-9. Controller—Descent Mode

If the LDG ALT is mistakenly set above the landing field elevation, the cabin will descend with the aircraft when aircraft altitude reaches the cabin altitude (Figure 12-10).

If the LDG ALT is mistakenly set below the landing field elevation, the aircraft will be pressurized at touchdown. The controller will ramp the cabin up at 500 fpm for 60 seconds after the squat switches signal weight-on-wheels. After 60 seconds, the controller will pull the outflow valves full open and the cabin will rapidly climb to field elevation (Figure 12-10).

MANUAL MODE

If the FAULT light illuminates, the controller has reverted to the manual mode. Depress the mode selector switch to turn out the FAULT and PRESS SYS lights. The MANUAL annunciator, in the MODE selector switch, will illuminate.

If the automatic mode of the controller is not operating satisfactorily, but the FAULT light has not illuminated, manual mode can be selected by depressing the MODE selector switch. The white MANUAL annunciator will illuminate.

Move the MAN ALT control UP, center, or DN and adjust MAN RATE control to maintain satisfactory pressurization in the manual mode.

If the cabin altitude exceeds approximately 8,750 feet while in the manual mode of operation, the PRESS SYS annunciator will illuminate. See Table 12-1 for a summary of events that will occur if the cabin climbs to abnormally high altitudes.

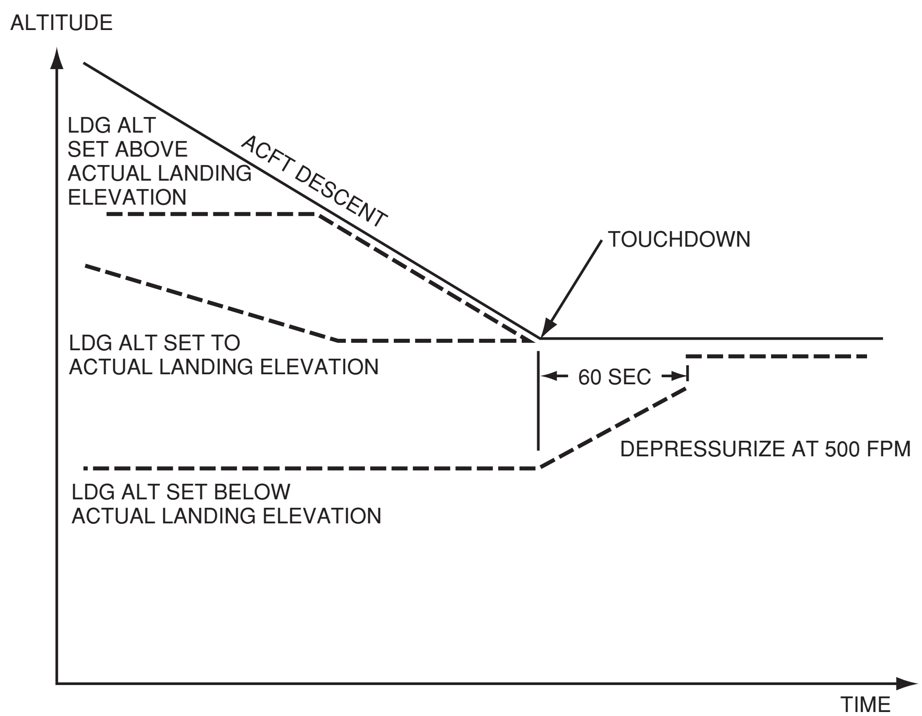


Figure 12-10. Controller—Landing Mode

HIGH ALTITUDE AIRPORT OPERATION

When discussing the pressurization system, airports above 8,000 feet are considered high altitude airports.

When landing the aircraft at a high field elevation, say 11,000 feet for example (Figure 12-11), it would be undesirable to have the controller fault to manual as the cabin climbed above 8,600 feet, emergency pressurization activate at 9,500 feet and cabin altitude horn activate at 10,100 feet. Therefore, the cabin controller is designed with a feature that resets (reindexes) all of these altitude-activated events to 14,500 feet when the following conditions exist: (1) the LDG ALT is set above 8,000 feet, (2) the aircraft has descended 1,000 feet, and (3) the aircraft has descended below 25,000 feet. When descent is initiated, the controller will start ramping the cabin altitude up, but will not exceed 8,000 feet until the aircraft has descended below 25,000 feet. Once below 25,000 feet, the controller will continue to ramp the cabin altitude up from 8,000 feet to whatever is selected on the LDG ALT scale.

See Table 12-2 for a summary of events that will occur if the cabin altitude reaches 14,500 feet when the conditions for landing at a high altitude airport exist.

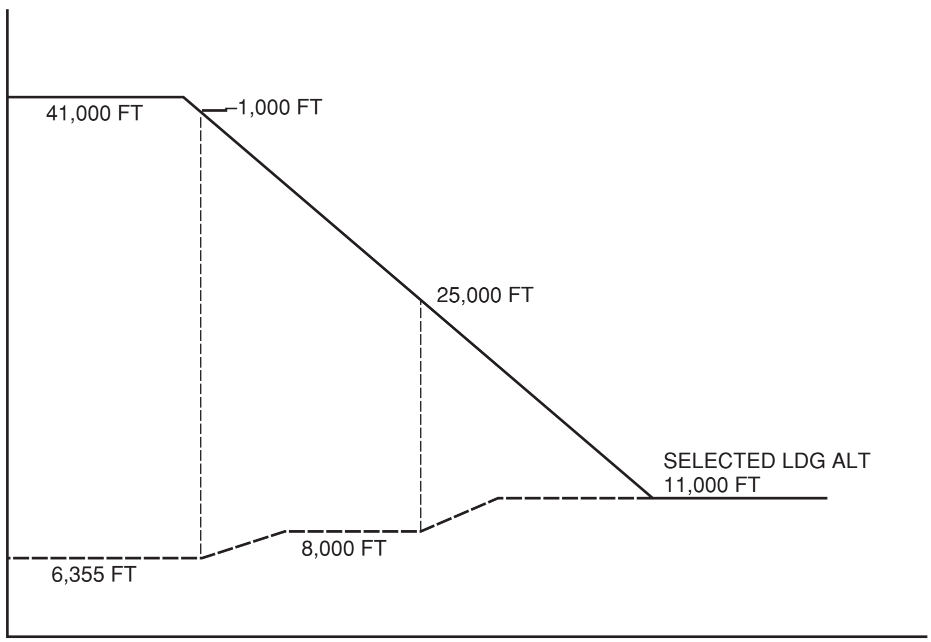


Figure 12-11. Controller—Landing Above 8,000 Feet

When takeoff is made from a field elevation above 8,000 feet, the cabin altitude will descend to and/or remain at 8,000 feet until the pressurization descent mode is entered (Figure 12-12).

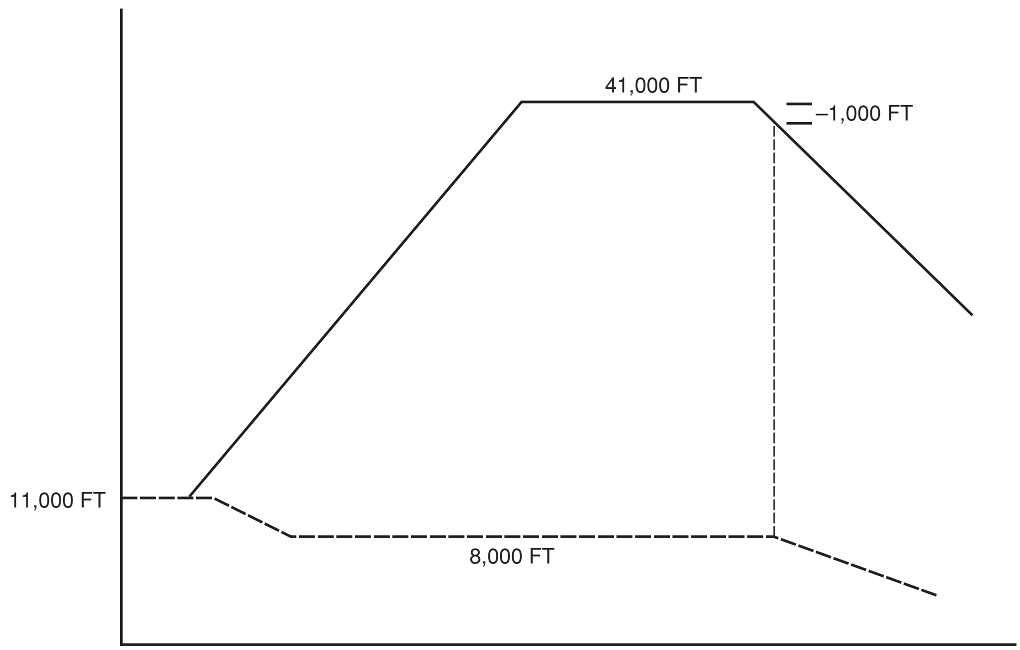


Figure 12-12. Controller—Takeoff Above 8,000 Feet

QUESTIONS

1. Which of the following valves does the cabin controller modulate to regulate cabin pressure?

A. Cabin safety valve
B. Flow control valve
C. Primary and secondary outflow valves
D. Primary and secondary differential pressure relief valves

2. What does illumination of the amber PRESS SYS light indicate?

A. Cabin altitude has exceeded 8,600 feet while operating in the automatic mode of operation.
B. The pressurization controller detects a fault.
C. Cabin altitude is at or above 9,500 feet and EMER PRESS mode has been activated.
D. Either A or B could be correct.

3. In the event of aircraft total electrical failure, which of the following statement(s) is/are true?

A. Cabin pressurization is controlled by the cabin pressure controller.
B. Cabin pressurization is controlled by the manual controller.
C. The emergency pressurization valves automatically actuate to provide cabin pressure.
D. Both B and C are correct.

4. At what cabin altitude does the cabin altitude warning horn sound?

A. 8,750 ±250 feet
B. 9,500 ±250 feet
C. 10,100 ±250 feet
D. 13,700 ±1,500 feet

5. Which of the following valves open at landing touchdown to dump residual cabin pressure?

A. Primary and secondary outflow valve
B. Cabin safety valve
C. Flow control valve
D. Both A and B

6. Illumination of the pressurization FAULT light could indicate:

A. Cabin altitude is above 8,600 feet.
B. Electrical power to cabin controller has failed.
C. Cabin controller has detected a fault.
D. Any of the above

7. What would cause the digits in the DIFF PRESS indicator to flash?

A. Differential pressure has exceeded 9.8 psid.
B. Differential pressure is zero.
C. Differential pressure has exceeded 9.9 psid.
D. DIFF PRESS indicator has detected an internal fault.

8. Which of the following statements is true regarding electrical power requirements for pressurization control?

A. 28 VDC is required for automatic and manual mode operation.
B. Pressurization controller will not operate without DC electrical power, but cabin pressure indicators will.
C. Automatic pressurization control is available in the EMER BUS mode of operation.
D. Manual mode operation and pressurization indicator are available without electrical power.

CHAPTER 13
HYDRAULIC POWER SYSTEMS

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CHAPTER 13
HYDRAULIC POWER SYSTEMS

INTRODUCTION

The hydraulic system is normally pressurized by two engine-driven pumps. An electrically-driven pump provides hydraulic pressure when an engine is not operating or when the main hydraulic system fails. The pressure for spoiler and thrust reverser operation can be provided only by the engine-driven pumps.

GENERAL

All three pumps in the hydraulic system draw MIL-H-5606 hydraulic fluid from a single reservoir that is pressurized to 17 psi by regulated engine LP bleed air. The engine-driven pumps are supplied through standpipes in the reservoir that limit the amount of fluid they can draw. Supply fluid flows through shutoff valves that can be closed from the cockpit in the event of fire or when maintenance is to be performed.

The DC motor-driven auxiliary pump, an integral part of the system, is supplied through a separate line from the bottom of the reservoir.

System pressure surges are dampened by two accumulators precharged with nitrogen. System pressure is displayed on an electrically powered indicator, and two annunciator lights warn of low pressure. The hydraulic pressure

indicator receives electrical power through the HYDRAULIC PRESS IND circuit breaker in the HYDRAULICS group of circuit breakers on the copilot's circuit-breaker panel. The hydraulic pressure indicator does not operate in the emergency bus mode of operation. The L and R HYDR PRESS annunciators are operable in the emergency bus mode of operation. The fluid output of all three pumps is filtered prior to reaching the sub-systems (landing gear, flaps, spoilers, thrust reversers, and brakes), and return fluid is filtered enroute to the reservoir. All return filters bypass at approximately 100 psi if elements become clogged. There is no cockpit indication of filter bypassing.

A system relief valve, set to relieve at approximately 1,750 psi, prevents excessive pressure in the system.

Hydraulic system servicing is not a pilot function; however, checking the pressure in the accumulators is a crew preflight item. A hydraulic ground service access below the right pylon allows viewing accumulator pressure. Precharge should normally be 850 psi (minimum of 750 psi) with system pressure on the cockpit hydraulic pressure indicator reading zero. The hydraulic reservoir, accessed through the tailcone door, incorporates a sight glass for checking fluid level. Conditions for checking fluid level should be: gear down, flaps and spoilers retracted, brakes released, and hydraulic pressure bled to zero. A floating silver ball should be at the top of the sight glass with the above conditions.

Controls and indicators for the hydraulic system are shown in Figure 13-1.

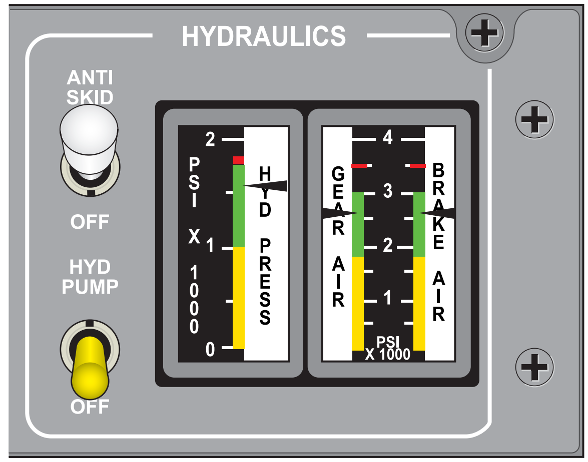


Figure 13-1. Hydraulic Control Panel

HYDRAULIC SYSTEM OPERATION

When DC electrical power is applied to the airplane prior to engine start, the amber L and R HYDR PRESS lights (Figure 13-2) will illuminate unless hydraulic pressure, at the respective engine pump, is above 150 psi.

Unless there is residual hydraulic system pressure, the auxiliary hydraulic pump must be operated to provide pressure for setting the parking brakes prior to engine start.

Placing the HYD PUMP switch (Figure 13-1) in the on (HYD PUMP) position starts the auxiliary pump, provided system pressure is below 1,000 psi. When the HYD PRESS indicator shows a pressure above 1,125 psi, a pressure switch stops the pump. When the pressure drops to 1,000 psi, the pressure switch will again start the pump. The auxiliary hydraulic pump is limited to three minutes on, followed by a 20 minute cooling period.

After starting the first engine, the HYD PUMP switch should be positioned to OFF, where it normally remains throughout the flight, unless required due to normal hydraulic system failure.

The supply lines to the 10-gpm, variable-volume, engine-driven pumps originate at standpipes in the reservoir. The standpipes limit the amount of fluid the pumps can draw, retaining the remaining fluid for the auxiliary pump in the event of a leak in the main hydraulic systems.

In the event of engine fire or when maintenance is to be performed, either of the normally open DC motor-operated shutoff valves can be closed by pulling the applicable ENG FIRE PULL T-handle on the glareshield. Pulling either T-handle also arms the fire extinguisher system; therefore, if valve closing is to facilitate maintenance, the applicable ENG FIRE EXT circuit breakers should be

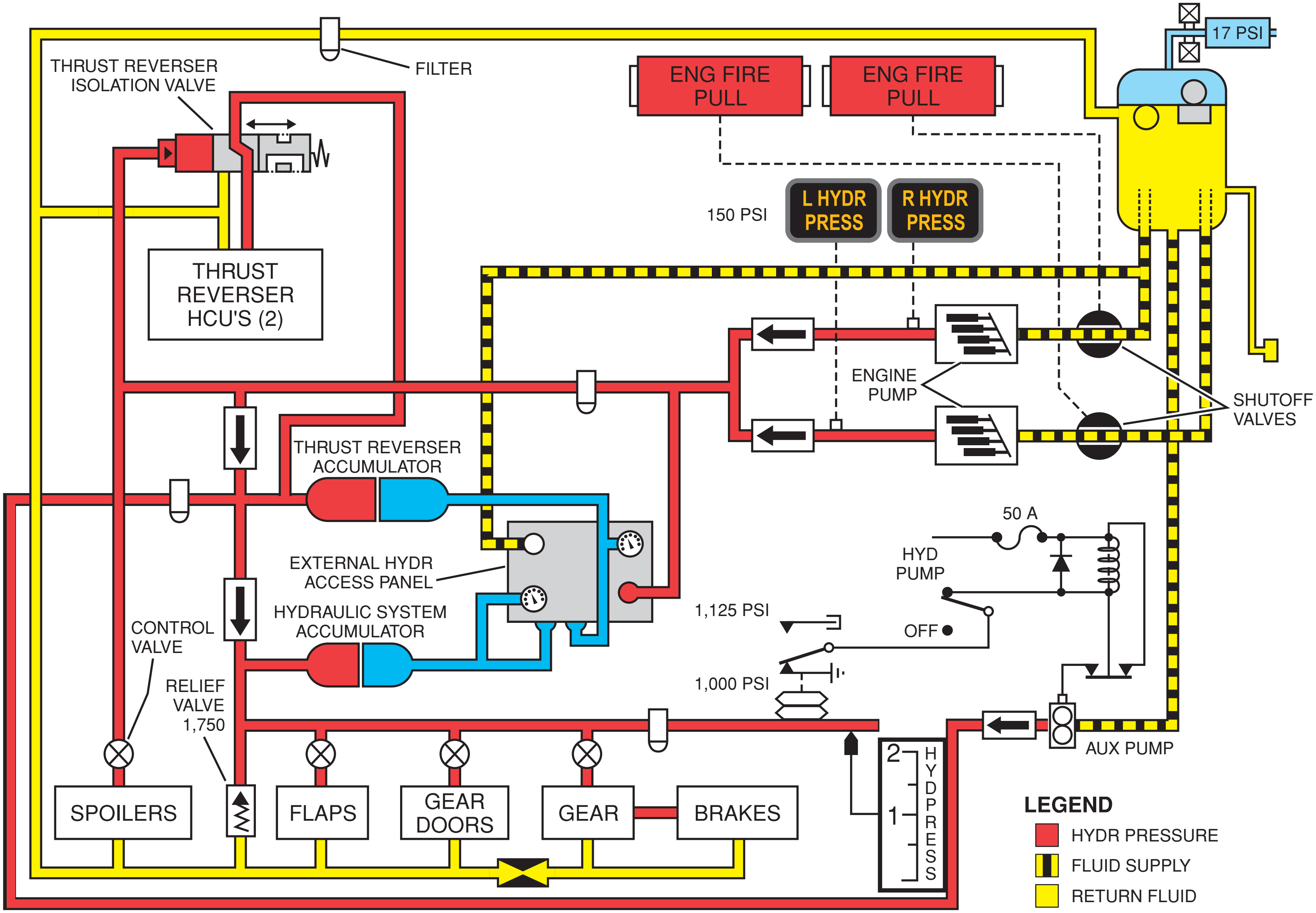


Figure 13-2. Hydraulic System

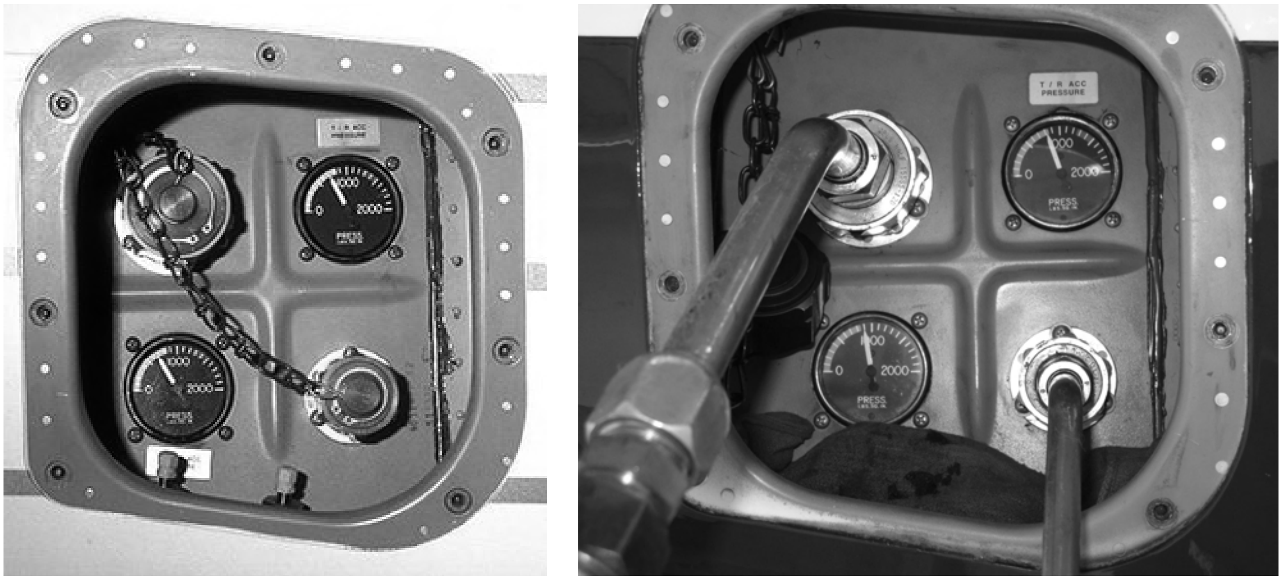


Figure 13-3. Hydraulic Service Access Panel

pulled to prevent accidental discharge of the bottles. The valves are opened by pushing in the applicable T-handle(s). The shutoff valves operate on 28 VDC supplies through the L and R FW SOV circuit breakers on the pilot's and copilot's circuit-breaker panels.

After starting the first engine, the HYD PRESS indicator should be checked to verify engine-driven pump operation. Pressure should stabilize at approximately 1,500 psi indicating that the engine-driven pump is operating properly.

When the second engine is started, there is no change in pressure indication, but flow capability is doubled. There is a positive indication that the second pump is operating properly, when the respective L or R HYDR PRESS light goes out (output pressure above 150 psi).

If an engine-driven pump fails in flight, the other engine-driven pump is capable of meeting system demands, but actuating speed may be reduced.

If all hydraulic system pressure is lost, the L and R HYDR PRESS light will illuminate as pressure decreases below 150 psi at the respective engine-driven pump. Do not operate the auxiliary pump until alternate landing gear extension procedures are executed as directed by the *AFM*.

HYDRAULIC POWER CART

A hydraulic power cart may be connected to the aircraft hydraulic system through connections on the hydraulic ground service access (Figure 13-3). This should be accomplished only by qualified personnel and in accordance with *Maintenance Manual* procedures.

After use of a hydraulic power cart, the airplane reservoir must be checked for proper hydraulic fluid level.

HYDRAULIC SUBSYSTEMS

Landing gear (including main gear doors), flaps, spoilers, thrust reversers and brakes are hydraulically powered. The application of hydraulic pressure is presented in Chapter 7—"Powerplant"; Chapter 14—"Landing Gear and Brakes"; and Chapter 15—"Flight Controls."

QUESTIONS

1. Normal hydraulic system pressure with the engine-driven pumps operating is:
A. 850 psi
B. 1,500 to 1,575 psi
C. 1,650 psi
D. 1,700 psi

2. The hydraulic shutoff valves are actuated:
A. By the ENG FIRE PULL T-handles
B. Automatically when the fire warning light comes on
C. By the generator off switch
D. By the bleed-air switches

3. In the event of hydraulic system pressure failure in flight:
A. Immediately position the HYD PUMP switch to ON.
B. Position the HYD PUMP switch to ON when the LO HYD PRESS light illuminates.
C. Refer to the hydraulic system failure Checklist.
D. Refer to HYDR PRESS light(s) illuminated in the Checklist.

4. In the event of hydraulic system failure, the L and R HYD PRESS light will illuminate at:
A. 150 psi
B. 1,500 psi
C. 1,250 psi
D. 850 psi

5. In the event of complete electrical failure:
A. The engine-driven pumps will continue to provide normal pressure, as shown on the HYD PRESS indicator.
B. Pressure will dissipate, and the auxiliary pump will have to be used for gear, flap, and brake operation.
C. The HYD PRESS indicator will show zero pressure, and the alternate gear extension system will have to be used to blowdown the landing gear.
D. The engine-driven pumps will continue to provide normal pressure, the HYD PRESS indicator will show zero pressure, but the landing gear can be extended normally.

6. The ________ subsystems cannot be actuated with auxiliary hydraulic pressure.
A. Landing gear
B. Spoiler and thrust reverser
C. Brakes
D. Flaps

7. The approved fluid for the hydraulic system is:
A. MIL-H-5606
B. MIL-O-M-332
C. MIL TYPE II
D. MIL-H-2380

8. The operational time limit of the auxiliary pump is:
A. 5 minutes on, 15 minutes off
B. 5 minutes on, 25 minutes off
C. 3 minutes on, 20 minutes off
D. 2 minutes on, 30 minutes off

9. The auxiliary hydraulic pump will provide:

A. 1,500 psi
B. 1,650 psi
C. 1,700 psi
D. 1,125 psi

10. If DC electrical power is applied to the airplane and residual hydraulic pressure is 1,300 psi:

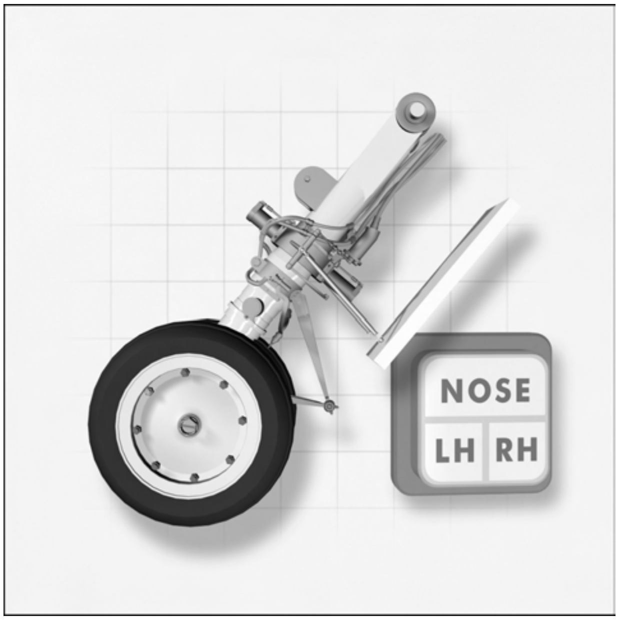
A. The auxiliary hydraulic pump will not operate when the HYD PUMP switch is on.
B. The L and R HYD PRESS lights will be out.
C. 1,300 psi will be shown on the HYD PRESS indicator.
D. A and C are correct.

CHAPTER 14
LANDING GEAR AND BRAKES

CONTENTS

ILLUSTRATIONS

CHAPTER 14
LANDING GEAR AND BRAKES



INTRODUCTION

The retractable landing gear is electrically controlled and hydraulically actuated. The main gear has dual wheels equipped with individual brakes and retracts inboard. The self-centering nose gear has a single wheel, incorporates electrical steering, and retracts forward. Alternate gear extension is pneumatic, backed up by free-fall capability. The hydraulic power brakes feature antiskid protection. Emergency braking is pneumatic.

GENERAL

The landing gear has three air-hydraulic type shock struts. The main gear has outboard doors that are mechanically linked to the gear and move with it. The inboard doors are hydraulically actuated and sequenced closed with the gear extended or retracted. An air bottle is provided for alternate gear blowdown. A separate air bottle supplies pressure for emergency braking and can also be used to initiate alternate gear free-fall. Gear actuators incorporate integral downlocking devices and no downlock pins are required. Gear position indications are displayed on the center instrument panel.

The power brake system is controlled by four power brake valves linked to the rudder pedals. Hydraulic pressure is metered to the self-adjusting, multiple disc brake assemblies in proportion to pedal deflection.

The antiskid system provides maximum braking without wheel skid. With the system selected, wheel speed sensors supply electrical signals to antiskid valves that modulate braking pressure. The parking brake is set with a handle on the center pedestal, using the same lines and brake assemblies as the normal braking system.

The variable-authority, digital nosewheel steering is operable only on the ground. When the system is engaged, rudder input signals, through the computer, determines the amount the DC-powered steering motor drives the nosewheel (explained later in this chapter). Maximum steering authority is 60° at slow speeds, decreasing at the higher taxi speeds as determined by main gear wheel speed transducer signals.

LANDING GEAR

INDICATING SYSTEM

General

The landing gear position indicating system consists of three red lights and three green lights, with dual bulbs, an amber MUTE switch, a test switch, and an aural warning horn.

Gear Position Lights

The three green DOWN lights (Figure 14-1) are down and locked indicators. As each gear locks down, the corresponding green DOWN light comes on. When gear retraction begins, the lights go out.

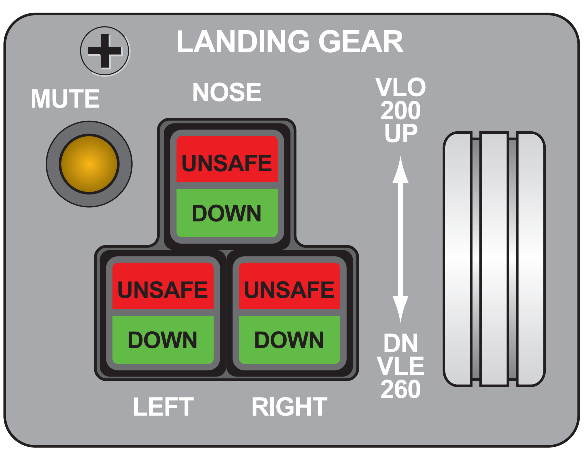


Figure 14-1. Gear Position Indicator Lights

The nose gear red UNSAFE light is on when the gear is in transit. When the nose gear is locked in either the up or the down position, the light goes out.

The two main gear red UNSAFE lights are wired to the two main gear inboard doors and are on when the gear is in transit since the inboard gear doors stay open until the main gear is either full up or full down. As each inboard door latches up, the corresponding red light goes out.

Indications for gear down and locked, up and locked, and in-transit conditions are shown in Figure 14-2.

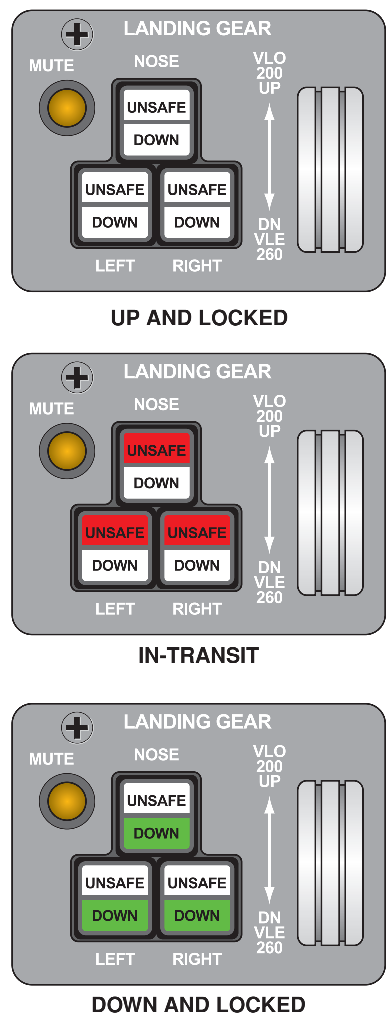


Figure 14-2. Gear Position Indicators

If the gear is extended with either alternate system, blowdown or free-fall, all three green lights and the two main gear red lights will be on (both inboard main gear doors will remain open).

The gear lights test with the main annunciator panel test, but the gear horn doesn't sound. Gear indicator lights and warning horn are tested by rotating the system test to the GEAR position and depressing the TEST button.

The RED/GREEN gear lights and amber MUTE light will illuminate and the warning horn will sound. Releasing the TEST button discontinues the test mode. The lights are automatically dimmed when the navigation lights are on.

Main gear green position lights circuitry and left and right land/taxi lights circuitry are routed through the main gear down and locked switches. If a main gear green indicator light fails to come on at extension, confirmation of gear down locking (after bulb testing with the SYSTEM/TEST switch) is made by switching on the applicable land/taxi light.

Nose gear green light circuitry is common with the nose steering system. Confirmation of nose gear down locking (after bulb testing) is made by positioning the nose steering switch on the pedestal to ON and observing that the ARM light, on the NOSE STEER switch, illuminates.

Landing Gear Warning System

The aural warning horn will sound and three red UNSAFE lights will come on when all the following conditions are present:

- Landing gear is not down and locked.
- Altitude is less than 16,300 feet.
- Airspeed is below 170 KIAS.
- At least one thrust lever is retarded below approximately 60% N_1.

When the horn sounds under the above conditions, it can be silenced by momentarily depressing the MUTE switch on the LANDING GEAR control panel position, or by depressing the MUTE button in the handle of the right thrust lever. The three red UNSAFE lights will not go out unless one of the conditions is corrected (airspeed must go above 190 knots for the lights to extinguish). The MUTE switch remains illuminated until the switch is disengaged or the above conditions are satisfied.

The warning horn will sound and cannot be silenced when the landing gear is not down and locked and the flaps are lowered beyond 25°. The red UNSAFE lights do not illuminate in this case.

MAIN GEAR COMPONENTS

Each main gear consists of a conventional air-hydraulic shock strut, dual wheels, scissor links, main gear actuator, inboard and outboard doors, and an inboard door actuator (Figure 14-3).

Figure 14-3. Main Gear Components

The main gear hydraulic actuator also serves as a side brace when the gear is extended. It features an integral downlock mechanism that can be unlocked only with hydraulic pressure on the retract side; therefore, no downlock pins are provided. Each main gear scissor links actuates a squat switch.

The main gear is hydraulically held in the retracted position and is enclosed by an outboard and an inboard door. The outboard door is mechanically linked to, and travels with, the gear.

The inboard door is hydraulically actuated, switch sequenced, and held retracted by hydraulic pressure and a spring-loaded uplatch which can be released by either of two actuators. The uplatches will hold the doors closed if hydraulic pressure is lost.

One of the actuators is hydraulically operated for normal extension and pneumatically operated for alternate extension (blowdown). The other actuator is pneumatically operated for free-fall extension.

Proper shock strut inflation is an important consideration. The shock struts are serviced to a specified pressure with the airplane on jacks. When the aircraft weight is on the struts, the amount of strut extension will vary with the aircraft load.

Squat Switch System

General

Some aircraft systems operate only on the ground while others operate only in the air. The squat switch system is designed to provide the necessary ground or airborne signals to these systems.

The squat switch system consists of two squat switches (Figure 14-4), one on each main landing gear strut scissors, and a relay box. When the airplane is on the ground and the main landing gear struts are compressed, the squat switches close to provide a ground mode signal.

When the aircraft lifts off the ground and the main landing gear struts extend, the squat switches open, interrupting the ground mode signals, thereby shifting to air mode.

Squat Switches

The squat switches provide ground or air signals to the following:

- FADEC
- Stall warning systems:

  The switches disable the stall warning test feature in the air.

Figure 14-4. Squat Switch

- Thrust reversers:

  Both squat switches must be in the ground mode for thrust reversers to deploy.
- Antiskid system:

  The switches disable the wheel brakes in the air with the antiskid system on. The wheel brakes remain inoperative until the wheels' spinup requirements have been met on landing.
- Gear control valve:

  The switches disable the gear-up solenoid on the ground to prevent inadvertent landing gear retraction. Either squat switch in ground mode will disable the gear-up solenoid. Both squat switches must be in the air mode to allow landing gear retraction.
- Autospoilers:

  Both squat switches must be in ground mode for autospoilers to extend (non spin-up aircraft only).
- Stabilizer heat:

  Disables stabilizer heat in the ground mode.

- APU
- Takeoff warning system
- Squat switch relay box:

  Either squat switch in the ground mode puts the relay box in ground mode. Both squat switches must go to air mode to put the relay box in air mode.

Squat Switch Relay Box

The squat switch relay box, like the squat switches, is either in the ground or airborne mode. If an electrical ground is available through either squat switch, the relay box is powered to the ground mode. The squat switch relay box uses DC power from the SQUAT SW circuit breaker in the TRIM–FLT CONT group on the left circuit-breaker panel to provide ground mode signals. With the SQUAT SW circuit breaker open, all the relay box functions go to air mode, even when on the ground.

NOTE

The position of the SQUAT SW circuit breaker has no effect on the landing gear, thrust reversers, antiskid, stall warning system operation or stabilizer heat. These systems receive signals directly from the squat switches, as explained previously.

Circuits to the squat switch relay box are:

- Nosewheel steering (ground operation)
- Cabin pressurization (airborne operation)
- Low-limit function of windshield heat and timer (ground operation)
- Mach trim test (ground operation)
- T.O. TRIM light for out-of-takeoff range (ground operation)
- SPOILER MON circuit cutout (ground operation)
- Generator current limiting (ground operation)
- Airstart relay (airborne operation)
- Cross–Start relay (ground operation)
- Stall warning lights and shakers (airborne operation)
- Fuel FILL–ON switch—The magnetic latch for the ON position of FILL-ON switch is released when the relay box goes to the air mode.
- Hourmeter (airborne operation), if installed
- Davtron Clocks
- TAT probe heating—Heating provided when in air mode
- Secondary Fuel Filter
- Weather Radar (standby on touchdown)
- FMS
- Master Caution Inhibit Function

Main Gear Wheel And Tires

Each main gear wheel incorporates two fusible plugs that prevent tire blowout caused by excessive heat resulting from hard braking. Tires are 17.50 X 5.78-8, 14 ply of the tubeless type (210 mph/182 knots rated), serviced to 210 psi loaded, or 202 psi on jacks.

NOSE GEAR COMPONENTS

The nose gear consists of an air-hydraulic shock strut incorporating a self-centering device, a nosewheel steering actuator, and mechanically operated doors (Figure 14-5).

The nose gear strut is conventional, with two exceptions: it does not have scissor links and the nosewheel steering actuator drive train is within the strut.

The nose gear actuator incorporates an integral downlock mechanism to maintain downlocked condition; therefore, a downlock pin is not provided. As with the main gear actuator, the locking mechanism can be released only by hydraulic pressure on the retract side.

The gear is held retracted by hydraulic pressure and a spring-loaded uplock hook that engages the uplatch roller on the forward

Figure 14-5. Nose Gear

side of the strut. The uplock hook holds the nose gear up if hydraulic pressure is lost. The uplock can be released by either of two actuators. One of the actuators is hydraulically operated for normal extension and pneumatically operated for alternate extension (blowdown). The other actuator is pneumatically operated for free-fall extension.

When retracted, the nose gear is enclosed by two doors that are linked to, and travel with, the gear.

An improperly centered nosewheel can jam in the wheel well; therefore, the nose strut incorporates a mechanism to center the wheel for retraction. At liftoff, two cams within the strut are engaged by strut air pressure, centering the wheel (Figure 14-6).

Since nosewheel centering depends on air pressure in the strut, proper inflation of the strut is especially important. As with the main gear, the nose strut is serviced to a specified pressure with the airplane on jacks.

When the aircraft weight is on the strut, the amount of strut extension will vary with airplane load.

Because the cams cannot center the wheel if it is swiveled 180° from the normal position, the nose gear should be checked, on the exterior inspection, to ascertain that the gear uplock roller is facing forward.

Nosewheel shimmy damping is provided through mechanical means and does not require electrical power.

Nose Gear Wheel And Tire

The nosewheel tire is chined to permit takeoff in up to three-quarters of an inch of water or slush. The chine deflects water or slush spray away from the engine intakes. The 18 X 4.4, 10 ply tire is rated to 210 mph (182 knots), serviced to 109 psi loaded or 105 psi on jacks.

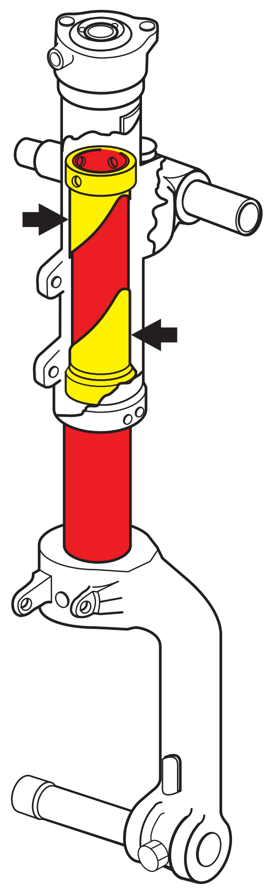


Figure 14-6. Nose Gear Centering Cams

OPERATION

The landing gear control valve is solenoid operated through switches sensing full open position of the main gear inboard doors. The circuit is routed through both squat switches to ensure that the airplane is off the ground prior to the valve being energized for retraction.

The gear door control valve is also solenoid operated. It is energized to the door open position when the landing gear selector switch is placed in either the UP or the DN position. It is energized to the door close position by gear-operated switches when the gear is fully retracted or extended.

Both control valves are pneumatically operated during landing gear alternate extension (blow-down or free-fall) and will not sequence the main gear doors closed.

Normal landing gear operation uses 28V (DC-1) power supplied through the GEAR circuit breaker on the copilot's circuit-breaker panel. The gear CB is on the EMER BUS in the event of electrical failure, the gear indicator lights are also powered by the No. 2 emergency battery, but gear operation is not. The gear would have to be extended using the alternate gear extension procedure with an electrical power failure.

Normal Retraction

Positioning the landing gear selector switch to UP energizes the door control valve, directing pressure to release the main gear inboard door uplatches and inboard door actuators to open the doors.

The two red main gear UNSAFE lights illuminate.

When the inboard doors are full open, door down switches on the inboard door actuators complete a circuit routed through both squat switches (air-mode) to energize the gear control valve, and hydraulic flow is directed to unlock and retract the landing gear (Figure 14-7). The three green DOWN lights extinguish and the red nose gear UNSAFE light illuminates.

When the gear has fully retracted, the red nose gear UNSAFE light extinguishes and switches are actuated that energize the door control valve. Pressure closes the gear inboard doors, which lock in position by spring tension of the door uplatches. The two main red UNSAFE lights extinguish.

Normal Extension

Positioning the landing gear selector switch to DN energizes the door control valve, directing pressure to release the main gear inboard door uplatch and to the door actuators to open the doors. The two red main gear UNSAFE lights illuminate.

When the inboard doors are full open, the gear control valve is energized and flow is directed to release the nose gear uplatch and to the gear actuators to extend the nose and main gear (Figure 14-8). The red nose gear UNSAFE light illuminates.

When the gear is fully down and locked, the three green down lights illuminate and the red nose gear UNSAFE light extinguishes. A circuit through the main gear downlock switches is completed to energize the door control valve. Pressure closes the gear inboard doors (Figure 14-8), which lock in position by spring tension of the door uplatches. The two red main gear UNSAFE lights extinguish.

Alternate Extension

General

There are two alternate gear extension systems, designated blowdown and free-fall. Two air bottles, charged to 1,800-3,000 psi, power the systems. Bottle pressures are shown on the GEAR AIR and BRAKE AIR indicators on the hydraulic control panel (Figure 14-9). The indicators operate on 28-VDC electrical power from the AIR PRESS IND circuit breaker in the HYDRAULIC group, on the right circuit-breaker panel.

The gear air bottle is used only for alternate gear extension-blowdown.

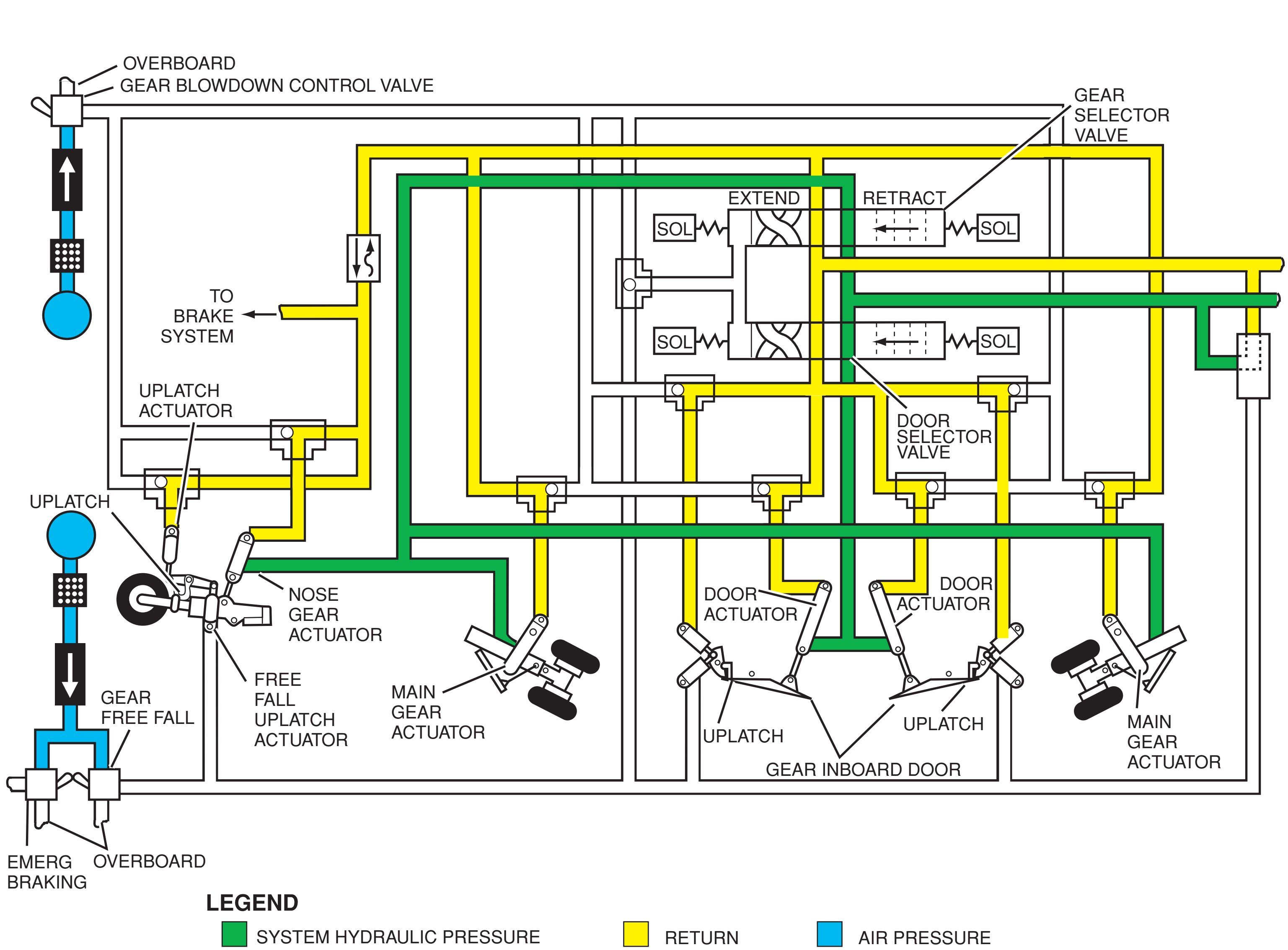


Figure 14-7. Landing Gear Retracted

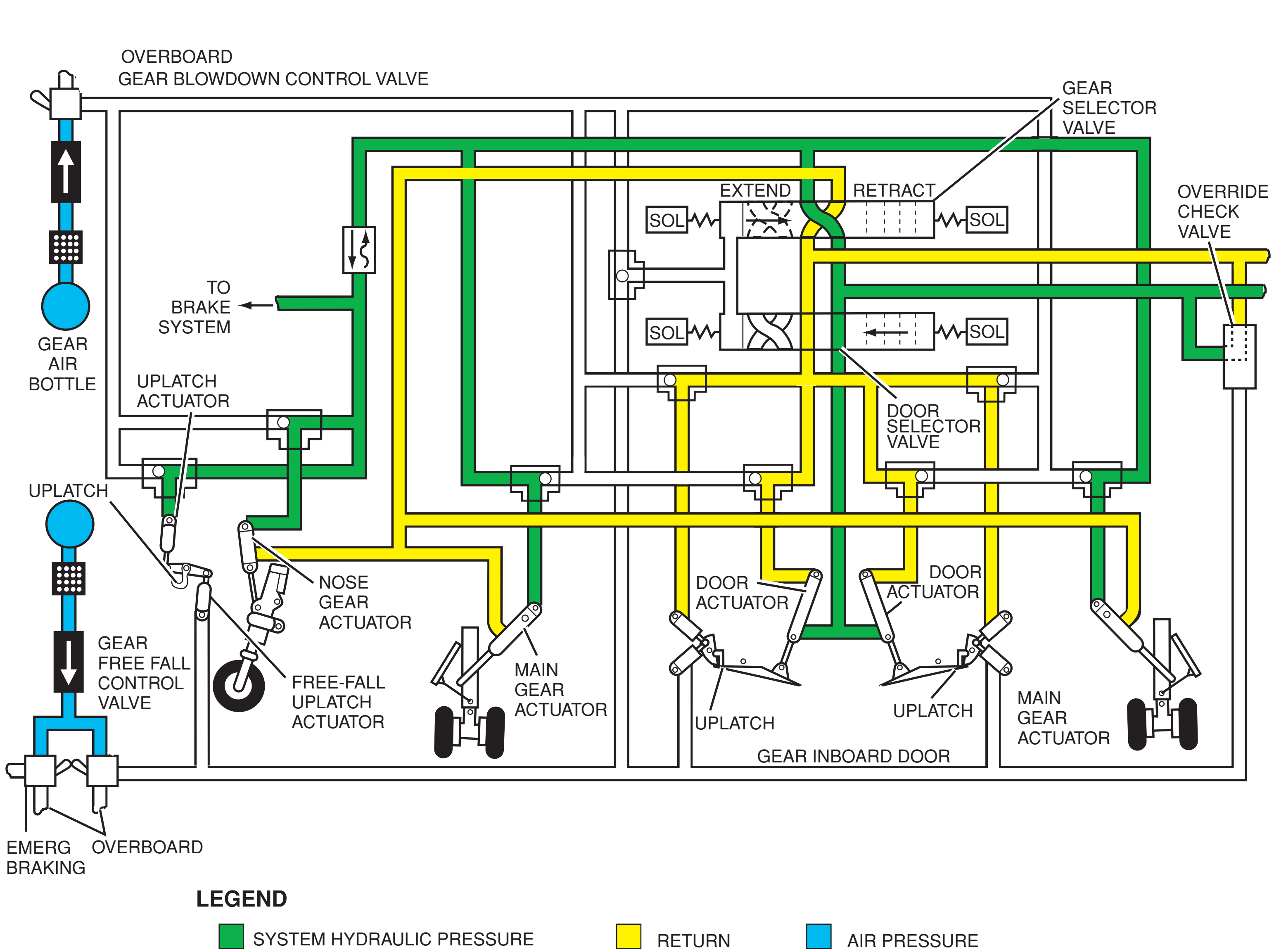


Figure 14-8. Landing Gear Extended

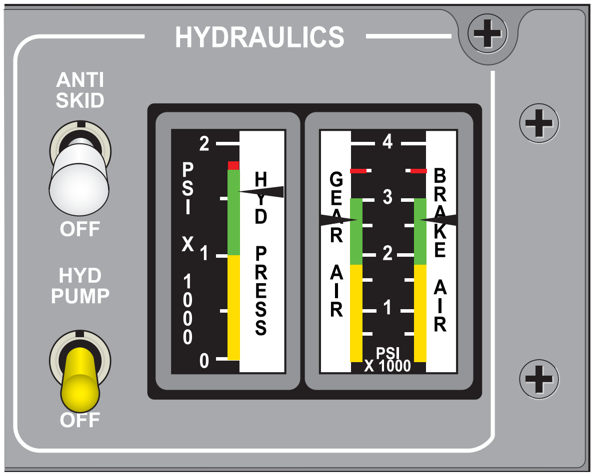
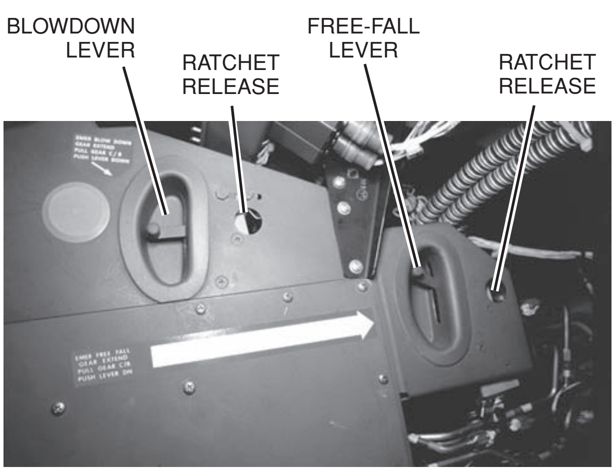


Figure 14-9. Air Pressure Indicators

The brake air bottle is primarily for emergency braking, but also provides pressure for the gear free-fall system. The blowdown system should be attempted first to conserve brake air for emergency braking (hydraulic failure), and to avoid interruption of main hydraulic system pressure (electrical malfunction). Prior to using either system, the landing gear selector switch (see Figure 14-2) should be placed DN and the GEAR circuit breaker on the copilot circuit-breaker panel should be pulled. This will prevent inadvertent gear retraction in the event electrical or hydraulic pressure is restored. The two systems are controlled by separate manual levers on the right side of the pedestal (Figure 14-10). Depressing either lever activates the corresponding system. The levers have ratchets to keep them locked in the down position, once activated, and can only be raised by actuating the ratchet release lever and simultaneously lifting the corresponding blow down or free-fall lever.



RIGHT SIDE OF PEDESTAL

Figure 14-10. Alternate Extension Levers

Blowdown

Pushing the BLOW DOWN lever to the full down and locked position actuates a valve to release GEAR AIR pressure to position the gear selector valve and the gear door selector valve to the extend position (Figure 14-10).

This prevents inadvertent gear retraction and provides a return flow path for fluid from the retract side of the gear and door actuators. If air pressure is greater than hydraulic pressure (electrical malfunction), the shuttle valves are repositioned and air pressure accomplishes the following:

- Releases the nose gear uplatch and the main gear door uplatches
- Opens the main gear inboard doors
- Extends all three gear

Both main inboard doors will remain open; therefore, the two main gear red UNSAFE lights will remain illuminated. The three green DOWN lights will illuminate.

The BLOW DOWN lever should not be repositioned to UP prior to landing. Do not attempt to retract landing gear after blowdown has been selected. To do so may cause excessive air pressure to be introduced into the hydraulic system return lines, thereby rupturing the reservoir (Figure 14-11).

Free-Fall

If the gear fails to extend with the blowdown system, revert to the free-fall system.

The free-fall system incorporates separate uplatch release actuators on the main gear inboard doors and the nose gear which are pneumatically actuated by the Brake Air bottle pressure when the free-fall system is activated. The system does not pressurize the gear actuators or gear inboard door actuators; how-

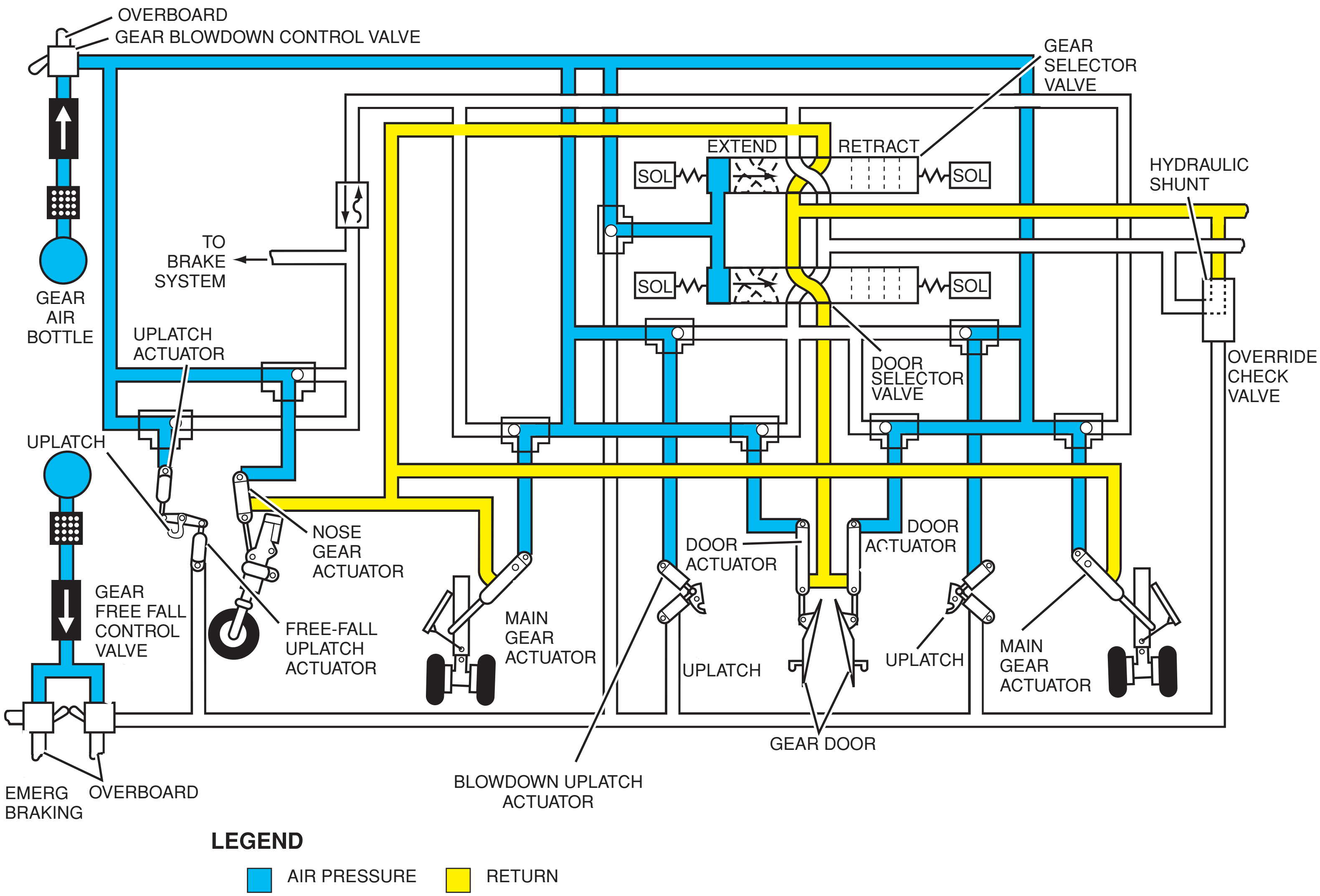


Figure 14-11. Alternate Gear Extension—Blowdown

ever, if blowdown was actuated first, pressure would still be applied to the actuators from the gear air bottle.

Pushing the FREE-FALL lever down actuates a valve to release Brake Air bottle pressure to position the gear selector valve and the door selector valve to extend position (Figure 14-10), if not previously positioned by the blowdown system.

Pressure is also directed to accomplish the following:

1. Release the nose gear uplatch and the main gear door uplatches.
2. Position the hydraulic pressure shunt (override check valve) to divert residual hydraulic system fluid pressure to return and deplete main hydraulic system pressure.

In addition to routing pressure (if any remains) to return, the hydraulic shunt (override check-valve) prevents system pressure from increasing. For this reason, the FREE-FALL lever should be returned to UP after using the system to prevent the shunt from releasing any hydraulic pressure that might be regained. Also, if the lever is left in the down position, brake air could be depleted if a leak exists in the free-fall system.

The gear and the inboard doors should free-fall to full extension. The green LOCKED DN lights should illuminate within 15 seconds for free-fall actuation if there is no residual hydraulic pressure.

As in blowdown extension, both inboard main gear doors remain open and the main gear UNSAFE lights remain illuminated (Figure 14-12).

BRAKES

GENERAL

The brakes can be applied by either the pilot or copilot. The system has multidisc, self-adjusting brake assemblies, one for each main gear wheel. Depressing the top section of the rudder pedals actuates the associated power brake valves which meter hydraulic pressure to the brakes. Braking force is in direct proportion to pedal application unless reduced by the antiskid system. An antiskid system, monitored by failure lights, permits stopping in the shortest possible distance under all runway conditions. Parking brakes can be set with a control handle on the center pedestal. A pneumatic emergency brake system is used to stop the airplane if hydraulic pressure is lost. Antiskid protection is not available during emergency braking (Figure 14-13).

The standard brake system utilizes a two rotor brake assembly. Aircraft modified by SB 60-32-12 utilize a three rotor brake assembly which provides for better braking performance and longer life. Due to the increased weight of the new brakes, the landing gear actuators were also redesigned to ensure sufficient force is available to retract the gear during takeoff.

OPERATION (WITHOUT ANTISKID)

Depressing either set of pedals opens the two brake valves and meters system hydraulic pressure (from the nose gear down line) sequentially through shuttle valves, open parking brake valves, antiskid valves, brake fuses (that close if a downstream leak occurs), and a second set of shuttle valves to all four brake assemblies (Figure 14-13).

Pistons in each brake assembly move a pressure plate, forcing the stationary and rotating disc together against a backing plate to produce braking action. Depressing one pedal applies both brakes on the corresponding main gear; therefore, differential braking is available, if required.

Releasing toe pressure on the pedals repositions the brake valve, and springs in the brake assembly return fluid in reverse flow through the brake valves to the reservoir, releasing the brakes.

A restrictor in the nose gear down line creates back pressure on the brakes as the landing gear retracts.

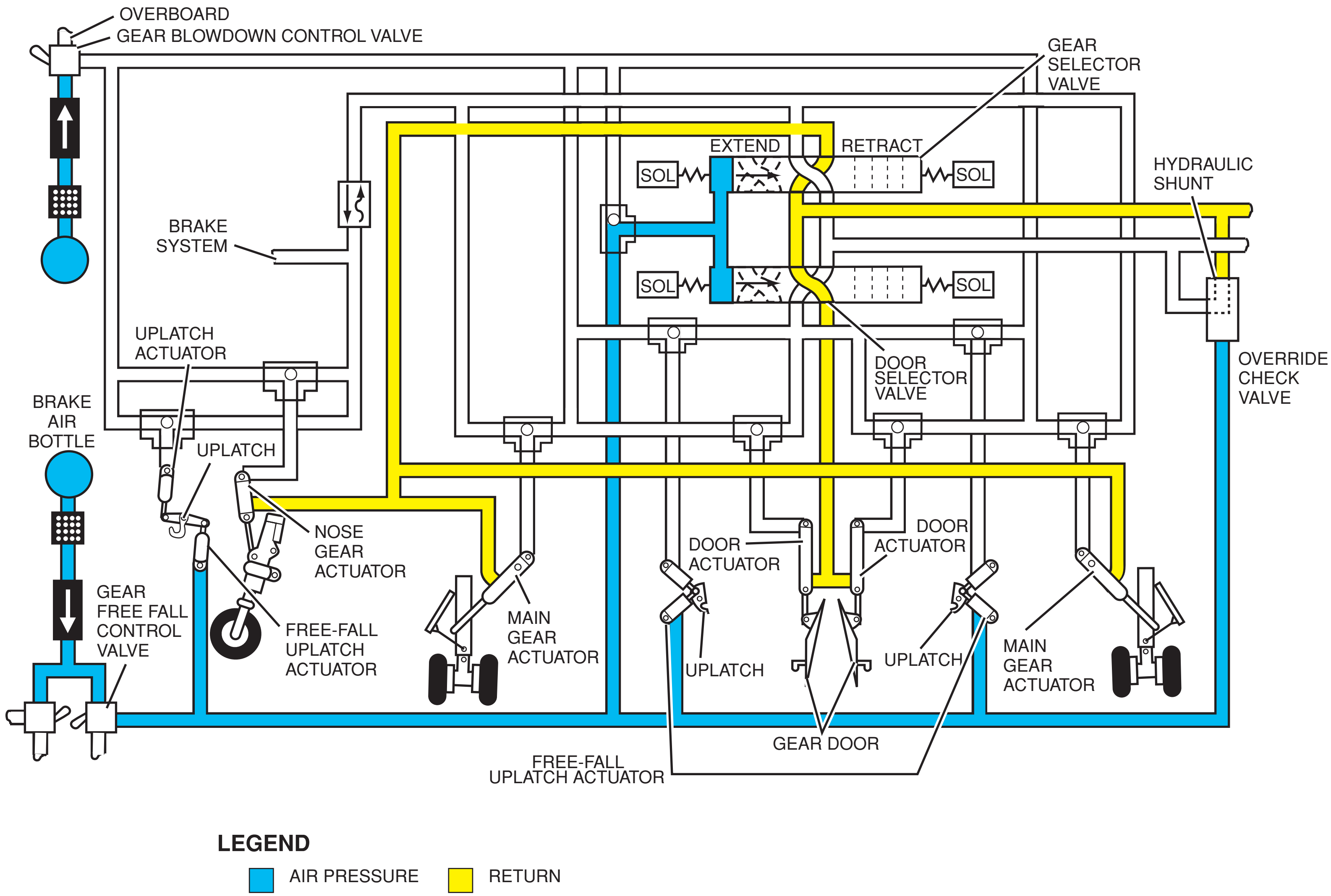


Figure 14-12. Alternate Gear Extension—Free-Fall

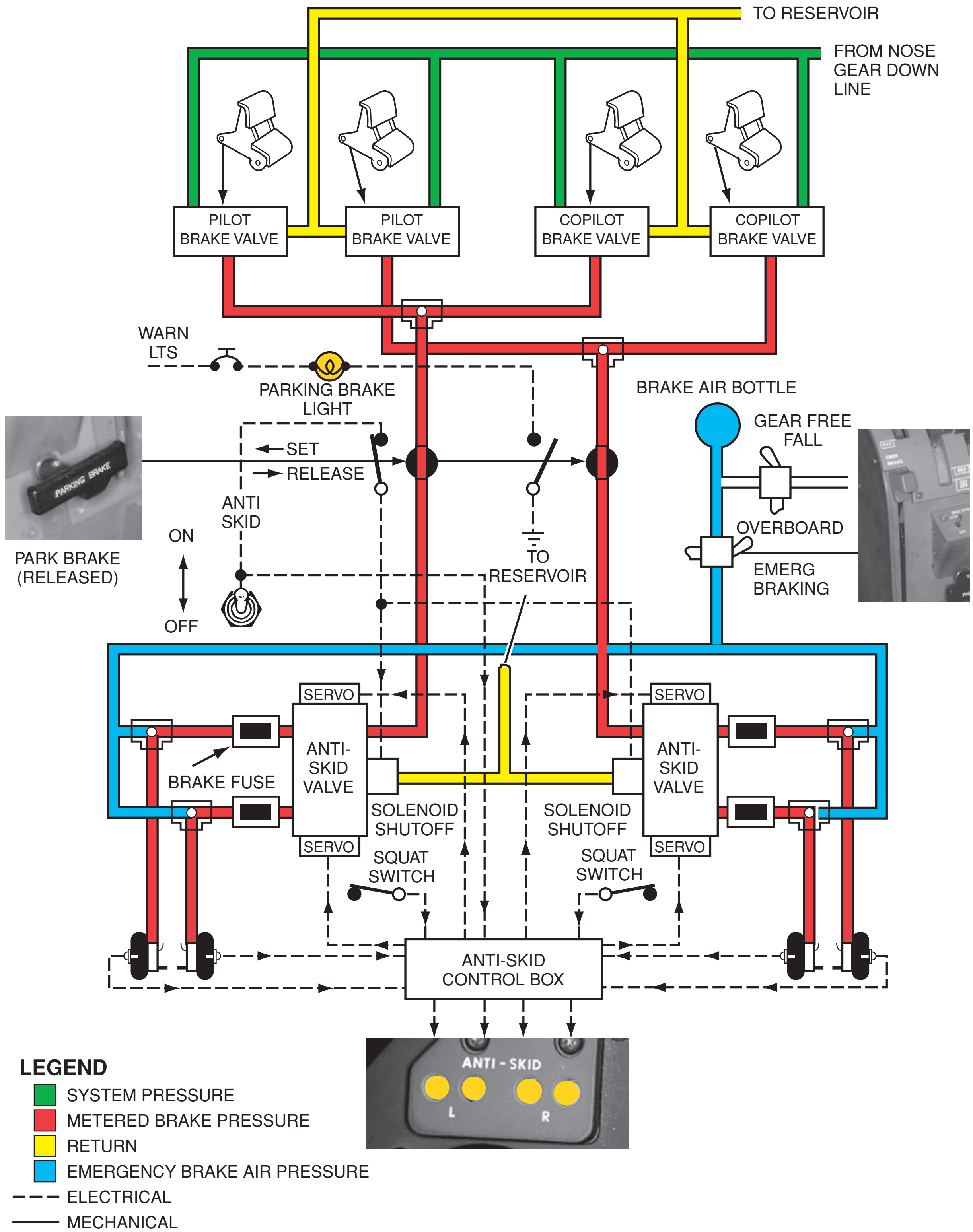


Figure 14-13. Brake System Schematic

This pressure is sufficient to stop the wheels from rotating prior to their entering the wheelwell.

When taxiing through slush or snow, brake application creates friction heat to prevent brake freezing

If a takeoff is made in slush or snow, brake assembly moving parts may freeze together after retraction. If frozen brakes are suspected after the gear is extended for landing, position the ANTISKID switch to OFF, apply the brakes 6 to 10 times to break up possible ice formations, then return the ANTISKID switch to ON. Operation of the wing anti-ice system will also aid in preventing frozen brakes, since bleed air is exhausted into the main gear wheelwell area when the system is operating.

OPERATION (ANTISKID)

General

The antiskid system controls braking on each main gear wheel independently, allowing maximum braking under all runway conditions without tire skidding.

The system consists of four wheel speed transducers (one on each main wheel), two antiskid control valves, a control box, monitor lights, and a lever-locking type ANTISKID switch on the hydraulic control panel.

The system is not required to be operative for flight. However, if one or more monitor lights are on, it must be assumed that the associated wheels do not have antiskid protection. In such a case, compute landing and takeoff distance and V_1 speed accordingly, and limit takeoff gross weight to 18,500 pounds.

Operation

With the ANTISKID switch on and the brakes applied, the control box receives and analyzes wheel speed inputs from the transducer on each main wheel (Figure 14-13). If the wheel deceleration rate is higher than a predetermined limit, the applicable control valve will individually regulate braking force on the corresponding brake by releasing braking pressure to the return line as required.

The ANTISKID switch is normally left on, although the system is inoperative at wheel speeds less than 160 rpm (approximately 10-11 knots).

To ensure full manual control of the brake system and to prevent pressure loss through the antiskid control valves, a solenoid-operated shutoff valve at each antiskid control valve return port is closed when the ANTISKID switch is off, or when the parking brake is set.

Four amber ANTISKID lights (Figure 14-13) monitor circuitry for each wheel, and will individually illuminate if a fault is detected. Cycling the ANTISKID switch off then back on may clear a malfunction. All four lights illuminate if power to the control box is lost, or if the ANTISKID switch is off and the landing gear is down. Electrical power for the antiskid system control circuits is 28 VDC supplied through the ANTISKID circuit breaker in the HYDRAULICS group of circuit breakers on the copilot circuit-breaker panel.

With the antiskid ON while the squat switches are in the airborne mode and the landing gear extended, these circuits prevent brake pressure from being applied to the brakes. If it is desired to exercise the brakes in flight (i.e., to remove ice), the ANTISKID switch must be turned OFF. The antiskid system is inoperative in emergency bus.

EMERGENCY BRAKES

The emergency brakes are to be used in the event of normal brake system failure. The possibility of wheel skids is increased when using this system since the antiskid system is not operative.

To apply brakes with the emergency system, pull the EMER BRAKE handle out of its recess (Figure 14-14) and press downward. The EMER BRAKE handle has to be pushed down approximately two inches before braking action begins. Modulate the handle while braking to get a feel for the system and to prevent wheel skid. Pressure from the brake air bottle is metered through the four shuttle valves to the brake assemblies when the spring-loaded handle is pressed down. Releasing the handle stops flow from the bottle and allows applied

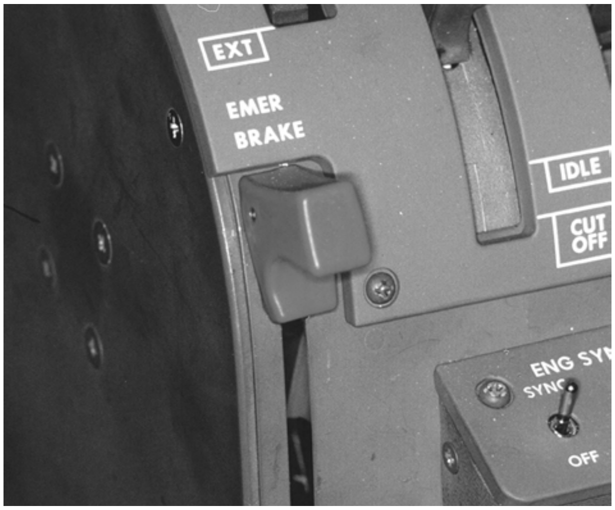


Figure 14-14. EMERG BRAKE Handle

air pressure to be vented overboard, releasing the brakes.

Since emergency braking applies all brakes simultaneously, differential braking is NOT available. Parking brakes are also inoperative during emergency braking.

Landing with a complete hydraulic failure or complete electrical failure (flaps, spoilers, and antiskid inoperative) will increase landing distance. See Dual Generator Failure or Hydraulic System Failure/Alternate Gear Extension in the *AFM* for procedure.

PARKING BRAKE

With an engine operating, normal hydraulic system pressure is used to set the parking brake. To set the parking brake prior to engine start, the auxiliary HYD PUMP switch must be placed to the ON position until system pressure is obtained, then positioned to OFF.

Apply brake pedal pressure and pull the PARKING BRAKE handle out. This closes the parking brake valve (see Figure 14-13), trapping applied pressure in the brakes. Pulling the handle also actuates a switch to close a solenoid-operated shutoff valve in each antiskid control valve, preventing inadvertent loss of brake pressure. A second switch is actuated when the parking brake handle is pulled. This switch causes an amber parking brake light above the antiskid light to illuminate.

NOTE

> A burned out parking brake annunciator light will cause the takeoff warning horn to sound when the squat switch relay box is in the ground mode and the right thrust lever is advanced above the MCT detent even if the parking brake handle is fully forward.

NOSEWHEEL STEERING

GENERAL

The digital nosewheel steering system is a computer controlled system that provides nosewheel deflection from 0° to a maximum 60° left or right for turning the aircraft when on the ground. The rudder pedals have limited nosewheel steering authority. Once the maximum rudder pedal displacement is reached, the pedals extend the range of nosewheel deflection by utilizing the generated force on the rudder pedals through force sensors.

The computer monitors ground speed, rudder pedal position, nosewheel position, and rudder pedal force to determine the correct nosewheel deflection. As the ground speed increases the maximum nosewheel deflection angle decreases. With a ground speed of 0 to 2 knots and maximum rudder force applied to the secondary stops, the wheel deflection is 60° left or right. At a ground speed of 85 knots the angle of deflection decreases to 0°. The torque on the nosewheel is faded to zero by 85 knots to allow smooth transitions between aerodynamic control and nosewheel actuator power loss during crosswind conditions. The system is disengaged when the ground speed increases to 90 knots and the green STEER ON annunciator light extinguishes.

System components consist of a nosewheel steering computer, steering actuator and gearbox, two position sensors (follow ups) on the gearbox, two rudder pedal follow ups me-

chanically connected to the pilot and copilot rudder pedals, and a ARM/NOSE STEER switch located to the left of the course heading control panel (Figures 14-15 and 14-16). The nosewheel steering computer also receives and processes signals from the yaw force interface box, the A/P electric box, the nose gear down and locked switch, and the squat switch relay panel.

When the NOSE STEER 28 VDC and 115 VAC circuit breakers in the TRIM FLT CONT group of circuit breakers on the right circuit-breaker panel are depressed, AC and DC is supplied to the nose steering computer. The computer in turn supplies 28 VDC to the nose strut steering actuator, 28 VDC to the yaw force interface box, (1 VAC) excitation voltage to the two nose strut position sensors (follow-ups), and 26 VAC to the two rudder pedal position sensors (follow-ups) when the system is engaged.

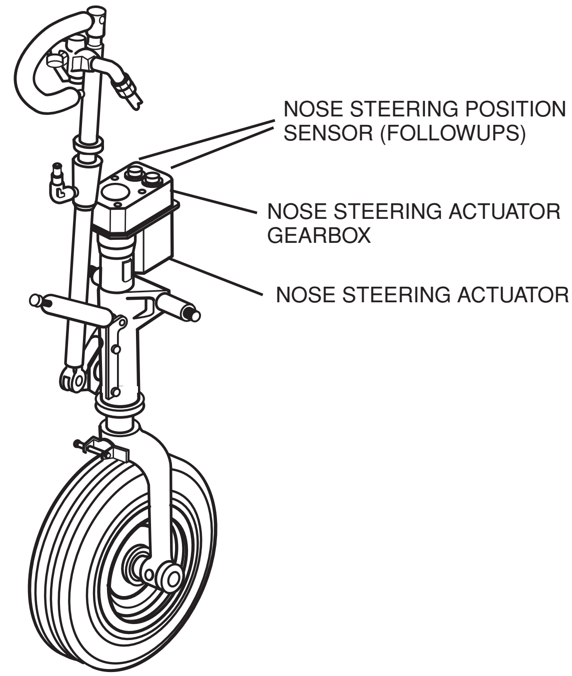


Figure 14-16. Nose Steering Components

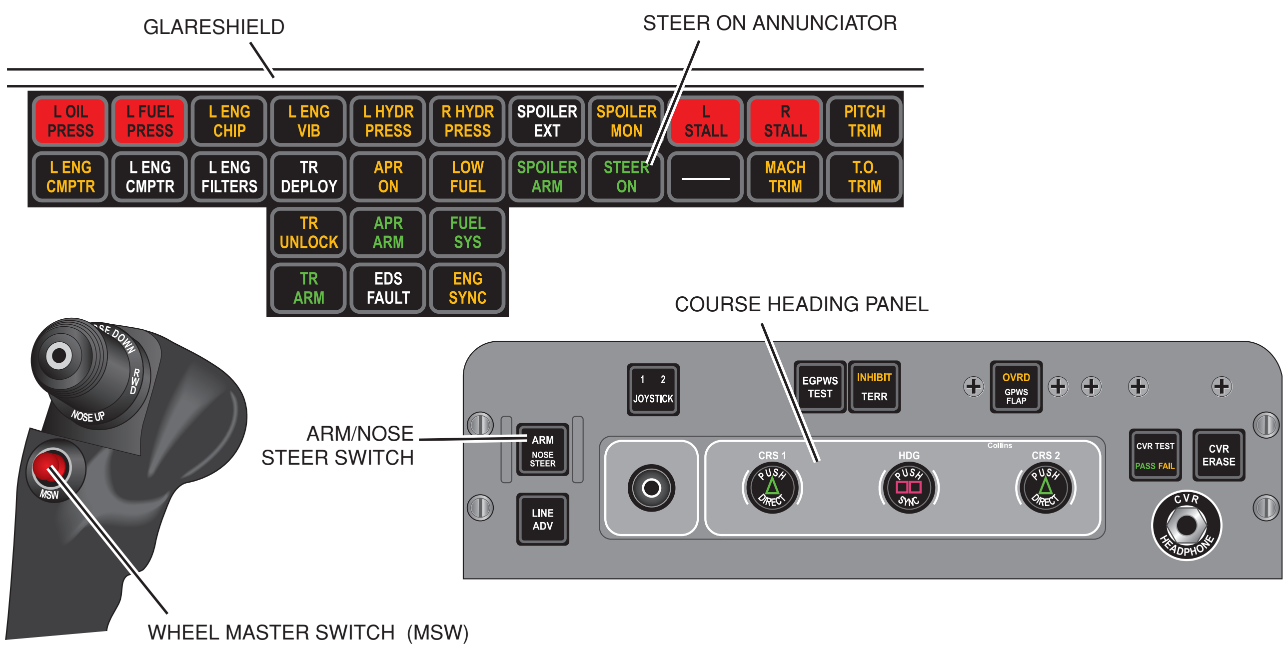


Figure 14-15. Nose Steering Controls and Indicators

OPERATION

Nosewheel steering can be operated by depressing and holding either control wheel master switch (MSW) or by momentarily depressing the ARM/NOSE STEER switch on the center pedestal. The green STEER ON annunciator illuminates when the steering is engaged in either mode.

The digital nosewheel steering system has two phases of operation when the ARM/NOSE STEER switch is depressed and released. In the first phase the computer is activated and in the second phase the system is engaged. Depressing and releasing the ARM/NOSE STEER switch on the center pedestal will initially activate the computer when a ground circuit from the nose gear down and locked switch is provided to the computer and no system faults are detected.

When the computer is activated, the ARM annunciator on the ARM/NOSE STEER switch shall illuminate and the green STEER ON annunciator (Figure 14-15) on the glareshield shall illuminate if the actuator has the capability to receive commands from the computer.

Once activated, the computer will remain activated until detection of a system fault, the ARM/NOSE STEER switch is depressed and released a second time (with nose gear down and locked only), or the pilot's or copilot's control wheel master switch is depressed and released (on ground only). When the computer is deactivated on the ground, the nosewheel steering disconnect tone will sound and the STEER ON annunciator extinguishes. When steering is engaged by holding the wheel master switch, it disengages when the master switch is released, and the green STEER ON light extinguishes, but the disconnect tone does not sound.

Retracting the landing gear while in flight will not deactivate the computer. The ARM annunciator on the ARM/NOSE STEER switch will extinguish but when the landing gear is lowered down and locked, the ARM annunciator on the ARM/NOSE STEER switch shall illuminate. Depressing the ARM/NOSE STEER switch has no effect on the system until the nose gear is down and locked.

The activated computer engages the system when a ground circuit from the squat switch relay panel is detected and the monitored ground speed is less than 90 knots. Once the system is engaged, 28 VDC is provided to activate the nose strut servo actuator, excitation voltage is provided to the nose steering position sensors, and to the rudder pedal position sensors.

The computer receives wheel speed information from the left hand and right hand inboard wheel speed transducers and the right hand outboard wheel speed transducer (Figure 14-17). The computer also receives rudder pedal information from the rudder pedal position sensors (follow-up), and rudder force information from the yaw force interface box. This information is processed and sent to the nose steering servo actuator as either a CW DRIVE or CCW DRIVE signal. The actuator then positions the nosewheel through the gear movement and sends this information back to the computer. The steering actuator (follow-ups) monitor gear movement and sends this information back to the computer. The computer corrects actual wheel deflection, as necessary.

The full input/range of the nosewheel steering variable authority is shown in Figure 14-18.

The nosewheel steering computer provides monitoring for protection against failures.

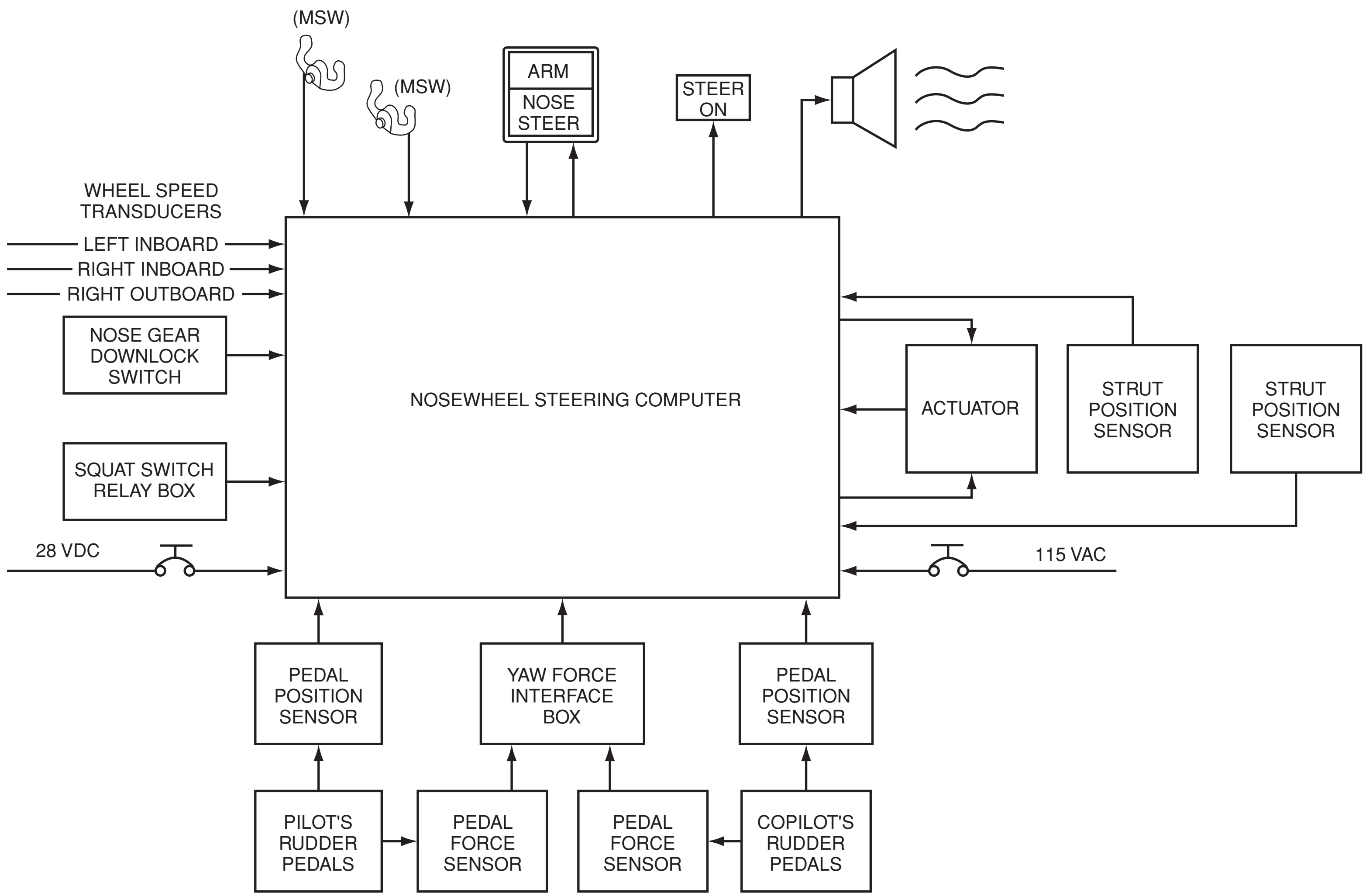


Figure 14-17. Digital Nosewheel Steering System

This monitoring is provided by processors in the digital computer. The computer monitors the following:

1. Wheel speed transducers on the main gear to determine ground speed.
2. Rudder pedal follow-ups to provide steering commands.
3. Rudder pedal force signals to provide steering commands.
4. Nosewheel follow ups to provide nosewheel displacement.
5. Ground speed is greater than 90 knots.
6. Nose gear is not down and locked.
7. Squat switches closed (on ground).
8. Servo is in the standoff condition. This occurs when the ground speed is at zero knots. The servo cannot drive the strut to the same position as the rudder pedals due to excessive friction. The computer limits the current of the servo to protect against burning the motor out with excessive current.

The digital nosewheel steering system will continue normal operation if one of the three wheel

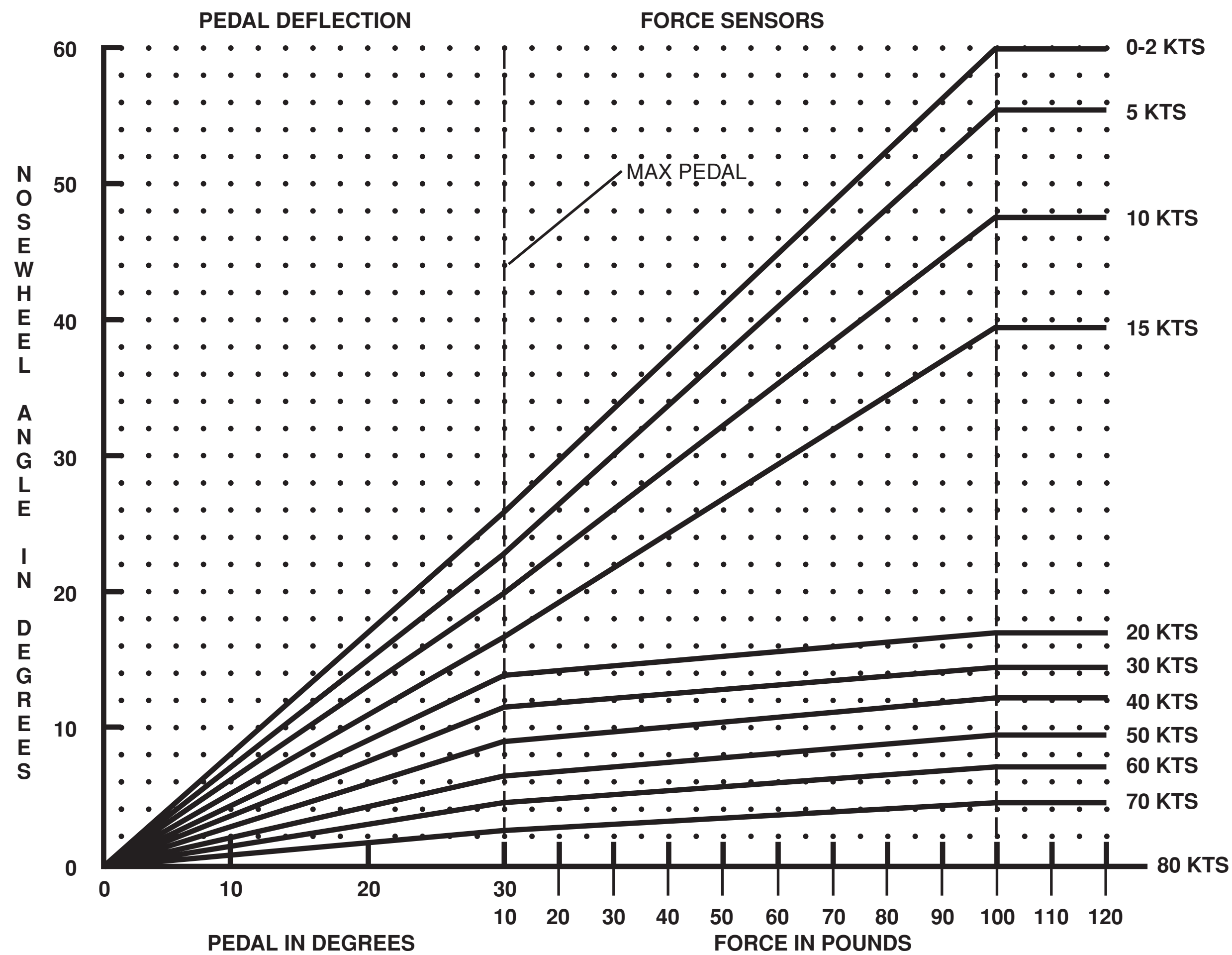


Figure 14-18. Nosewheel Steering Variable Authority

speed signals is lost. Once disarmed, the system shall not rearm until the fault is corrected (STEER ON and ARM lights extinguished).

Loss of two wheel speed signals will disengage the system. Once disengaged, it will not rearm to alert of an internal failure. In this case, the nosewheel steering can still be operated through the control wheel master switch up to 10 knots. The disconnect tone will not sound at the 10-knot point.

Loss of one of the pedal force signals will disarm the system. Limited nosewheel deflection up to 24° can be achieved by depressing and holding either the pilot or copilot control wheel master switch.

NOTES

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QUESTIONS

1. Emergency air pressure can be used for:

A. Gear extension and parking brake
B. Gear, flaps, spoilers, and brakes
C. Gear extension and brakes
D. Gear extension, flaps, and brakes

2. GEAR AIR and BRAKE AIR pressure indicators should indicate ______ prior to takeoff.

A. 1,800 to 3,000 psi
B. Minimum 1,700 psi
C. 3,000 to 3,350 psi
D. Maximum 1,750 psi

3. During ***normal*** gear operation, main gear inboard doors and the main gear are sequenced by:

A. Micro switches
B. Emergency air pressure
C. Mechanical link
D. Both A and B

4. After the gear is fully retracted, the gear position light indications will be:

A. Three green DOWN lights if the throttles are below 60% to 70% N_1.
B. Three green DOWN lights
C. Three red UNSAFE lights until above approximately 15,000 feet
D. None of the lights should be illuminated if throttles are approximately 60% or more.

5. After an alternate gear extension, the gear position light indication will be:

A. Three green
B. Three green, two red
C. Three red, two green
D. Three red, three green

6. Three gear UNSAFE lights will be on and the gear warning horn sounding when the:

A. Gear is retracted and no green DOWN lights are on.
B. Gear is down and throttles are above 60% N_1 below 15,300 feet.
C. Gear is up and the throttles are below 60% N_1, below 170 KIAS and below 16,300 feet.
D. Flaps are extended below 25 degrees regardless of altitude.

7. The gear warning horn will sound without any gear light indications when the:

A. Gear is up the throttles are below 60% - 70% N_1 below 14,500 feet.
B. Gear is up and flaps are below 25°.
C. Gear is down and V_{LE} is exceeded.
D. Gear is up, spoilers are extended, and flaps are below 3°.

8. Illumination of the main gear red UNSAFE lights with gear down and three green lights indicates:

A. The corresponding main gear is not down and locked.
B. The corresponding main gear is not up and locked.
C. The corresponding main gear inboard door is not fully closed.
D. The corresponding main gear inboard door is locked in the closed position.

9. The nose gear red UNSAFE light will be on when:

A. The nose gear is unsafe or in transit.
B. Nosewheel steering is inoperative.
C. The nose gear doors are open.
D. The nose gear doors are closed.

10. To set the parking brake:

A. Hydraulic system pressure is not required.

B. Hydraulic system pressure is required.

C. Only the pilot's brake pedals can be used.

D. Antiskid must be off.

11. When antiskid is inoperative:

A. The airplane may not be dispatched.

B. Landing/takeoff distances and V_1 speed will be affected.

C. Takeoff V_R will be affected.

D. B and C are both correct.

12. Normal brake pressure is provided by:

A. Main hydraulic system through the nose gear down line

B. Brake accumulator

C. Brake air bottle through the antiskid control valves

D. Gear air bottle

13. If a green DOWN light is burned out, main gear down and locked condition can be confirmed by:

A. GND IDLE light illuminated

B. ENG SYNC light illuminated

C. Illumination of the corresponding landing light

D. Red UNSAFE lights illuminate

14. The electrical requirements for nosewheel steering are:

A. 24 VAC and 28 VDC

B. Only 28 VDC

C. Only 115 VAC

D. 28 VDC and 115 VAC

15. When the antiskid system is operating properly, all the ANTI-SKID lights:

A. Will be extinguished, provided the ANTI-SKID switch is on

B. Will be on, provided the ANTI-SKID switch is on

C. Will be on when the parking brake is set

D. Will be on when wheel speed is less than 150 rpm

16. The nosewheel steering actuator is a:

A. 28 VDC motor

B. 115 VAC motor

C. Hydraulic motor

D. Vane-type hydraulic actuator

CHAPTER 15
FLIGHT CONTROLS

CONTENTS

ILLUSTRATIONS

CHAPTER 15
FLIGHT CONTROLS

INTRODUCTION

The manually actuated primary flight controls incorporate electrical trim in all three axes. Secondary flight controls consist of hydraulically actuated spoilers/spoilerons and flaps.

Other systems related to flight controls are yaw damper, rudder boost, stall warning, Mach overspeed/warning, and Mach trim.

GENERAL

The primary flight controls (ailerons, elevator, and rudder) are mechanically operated through the dual-control columns, control wheels, and rudder pedals.

The ailerons incorporate mechanical balance tabs to provide aerodynamic assistance. Trim systems (pitch, roll, and yaw) are electrically operated and controlled. The left aileron and the rudder feature trim tabs. The movable horizontal stabilizer provides trim in the pitch axis.

All flight control trim motors and electrical servos can be disabled by depressing and holding either control wheel master switch (MSW) in the event of a malfunction causing uncommanded control inputs.

The flaps, spoilers, and spoilerons are hydraulically operated and electrically controlled. All flight control surfaces are shown in Figure 15-1. Aileron augmentation is provided by a spoileron system to increase roll capability when the aircraft is configured for landing (DN FLAPS).

A Mach trim system provides automatic pitch trim to compensate for a slight reduction in stick force gradient in the transonic region.

A controls gust lock is provided to prevent wind gust damage to the primary flight control surfaces. When installed as depicted in Figure 15-1, the lock holds full left rudder, full up left aileron, and full down elevator displacement.

PRIMARY FLIGHT CONTROLS

ELEVATORS

The elevators (Figure 15-2) are hinged to the aft edge of the horizontal stabilizer and are connected through cables, push/pull tubes, sectors, and bellcranks to the control columns. They are positioned through this mechanical linkage by fore-and-aft movement of the control column. Three scuppers are located near the aft edge of each elevator for moisture drainage, and three static wicks are attached to the trailing edge of each elevator.

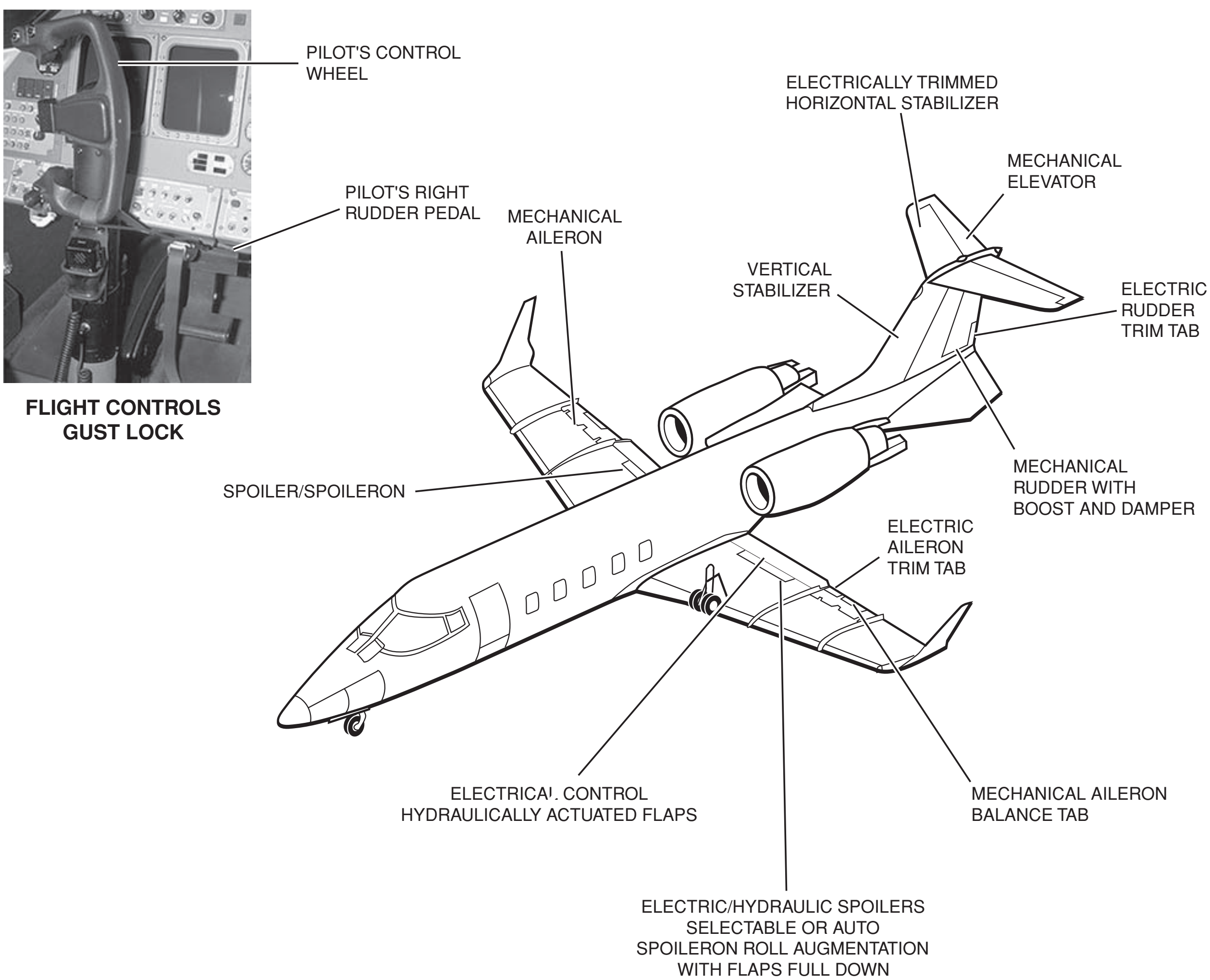


Figure 15-1. Flight Control Surfaces

The elevators can be electrically positioned through the autopilot pitch servo in addition to manual operation through the control column movement.

Pitch Servo

The airplane has one pitch servo actuator for use by the autopilot (Figure 15-2). Autopilot operation is described in Chapter 16, Avionics.

The pitch servo actuator is a two-direction torque motor; it incorporates an electrical clutch that is disengaged unless the autopilot is connected to utilize the servo. If a system malfunction causes the servo to produce an undesired elevator movement, the crew can disengage the servo clutch by depressing and holding the wheel master switch (MSW) on either control wheel. The pilot can also manually override the pitch servo, if necessary.

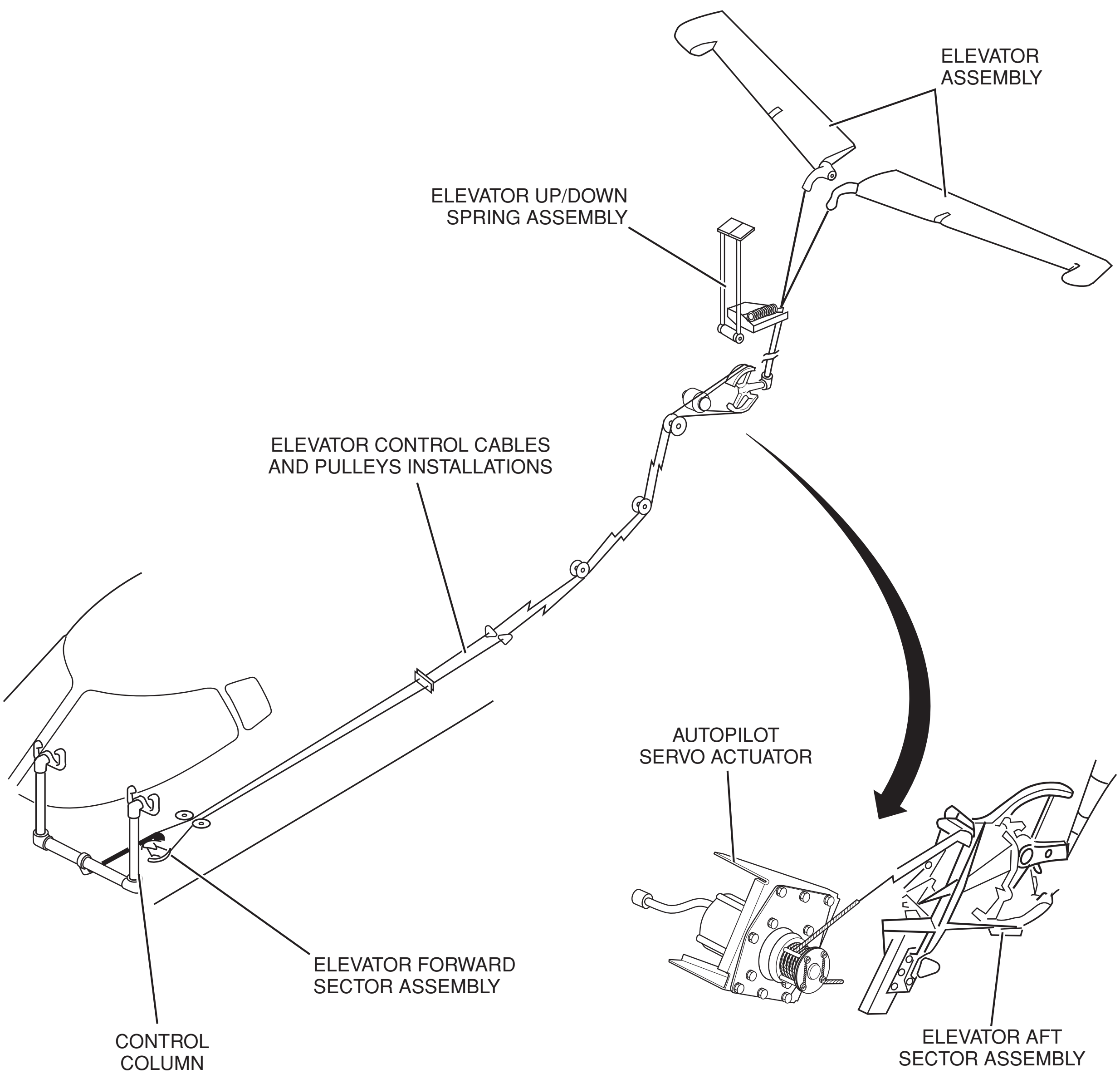


Figure 15-2. Elevator Control System

The PITCH SERVO circuit breaker, in the AFCS group of circuit breakers on the pilot circuit-breaker panel, powers the DC servo.

Up/Down Spring

An up/down spring assembly in the elevator control linkage improves pitch stability.

AILERONS

The ailerons (Figure 15-3) are hinged to the trailing edge of the wings and are connected through pulleys and cables to the control wheels. The ailerons can be mechanically positioned with either control wheel or by an autopilot roll servo actuator to provide control in the roll axis. Aileron effectiveness is augmented by spoilerons when the aircraft is configured for landing (FLAPS FULL DOWN).

Spoileron (aileron augmentation) operation is automatically activated when the flaps are lowered below 25°. In the spoileron mode, when an aileron is moved up to initiate aircraft roll, the spoiler on the same wing automatically raises the same number of degrees (15° maximum) to provide additional roll control.

Roll Servo (Autopilot)

The ailerons can also be positioned by the autopilot roll servo. The servo incorporates an electrical clutch that engages only when the autopilot roll axis is engaged. In the event of a malfunction, the servo can be manually overridden by the pilot or disconnected by disengaging the autopilot. The 28 VDC ROLL-YAW SERVO circuit breaker is in the AFCS group of circuit breakers on the pilot circuit-breaker panel.

Balance Tab

The mechanical balance tab on each aileron (Figure 15-3) moves proportionally in the opposite direction of the aileron and provides aerodynamic assistance in moving the aileron against airloads, thus reducing control wheel force with increasing airspeed.

Trim Tab

The aileron trim tab is attached to the inboard trailing edge of the left aileron (Figure 15-3). The tab is positioned by either control wheel trim switch. Aileron trim tab position is indicated on the cockpit forward instrument panel. Uncommanded aileron trim can be stopped by depressing and holding either control wheel master switch (MSW).

Aileron Follow-ups

Aileron follow-up mechanisms, driven by aileron control linkage, provide aileron displacement information to the spoileron computer and to the autopilot.

RUDDER

The rudder (Figure 15-4) is hinged to the aft edge of the vertical stabilizer and is connected through a sector, drive, cables, and rudder pedal linkage to both sets of rudder pedals. The rudder can be manually positioned with either set of rudder pedals or by the yaw damper servo. The crew can manually override the yaw damper in the event of a malfunction or disengage it by depressing either wheel master switch or the yaw damper YD button on the autopilot control panel (Figure 15-4).

The yaw damper servo receives 28 VDC power through the ROLL-YAW SERVO circuit breaker in the AFCS group of circuit breakers on the pilot's circuit-breaker panel.

Rudder Boost System

The rudder boost system reduces rudder forces, increases directional control effectiveness, and improves takeoff performance especially during single engine operations. Whenever the flaps are 8° or lower, and the pilot exerts approximately 50 pounds of rudder pedal force, the system engages (if armed) and uses the yaw damper servo to assist the pilot in deflecting the rudder. A RUDDER BOOST switch is used to arm the system and system status is provided by green RB (active) and amber RB (fail) lights on the glareshield

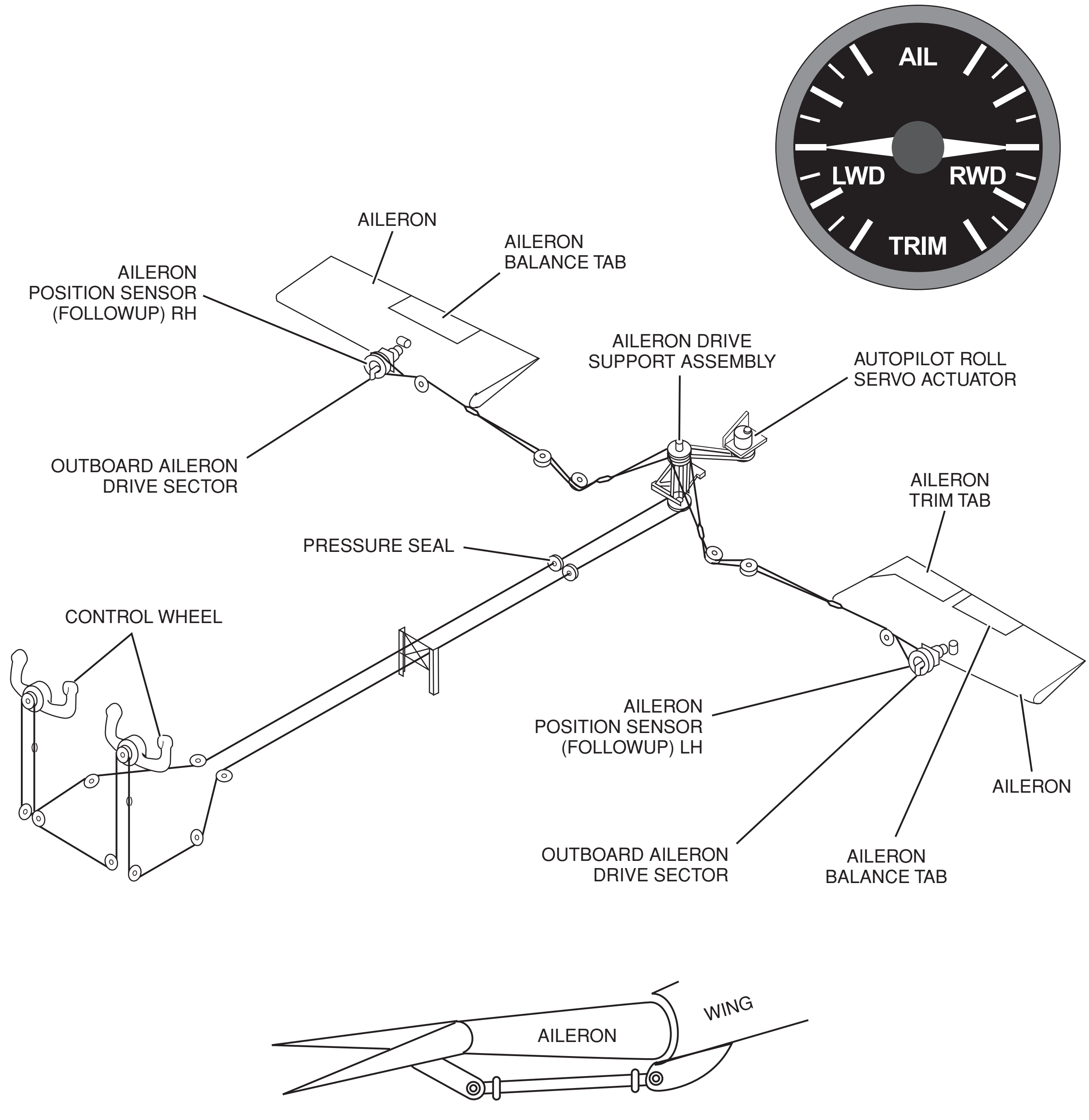


Figure 15-3. Aileron Control System

15 FLIGHT CONTROLS

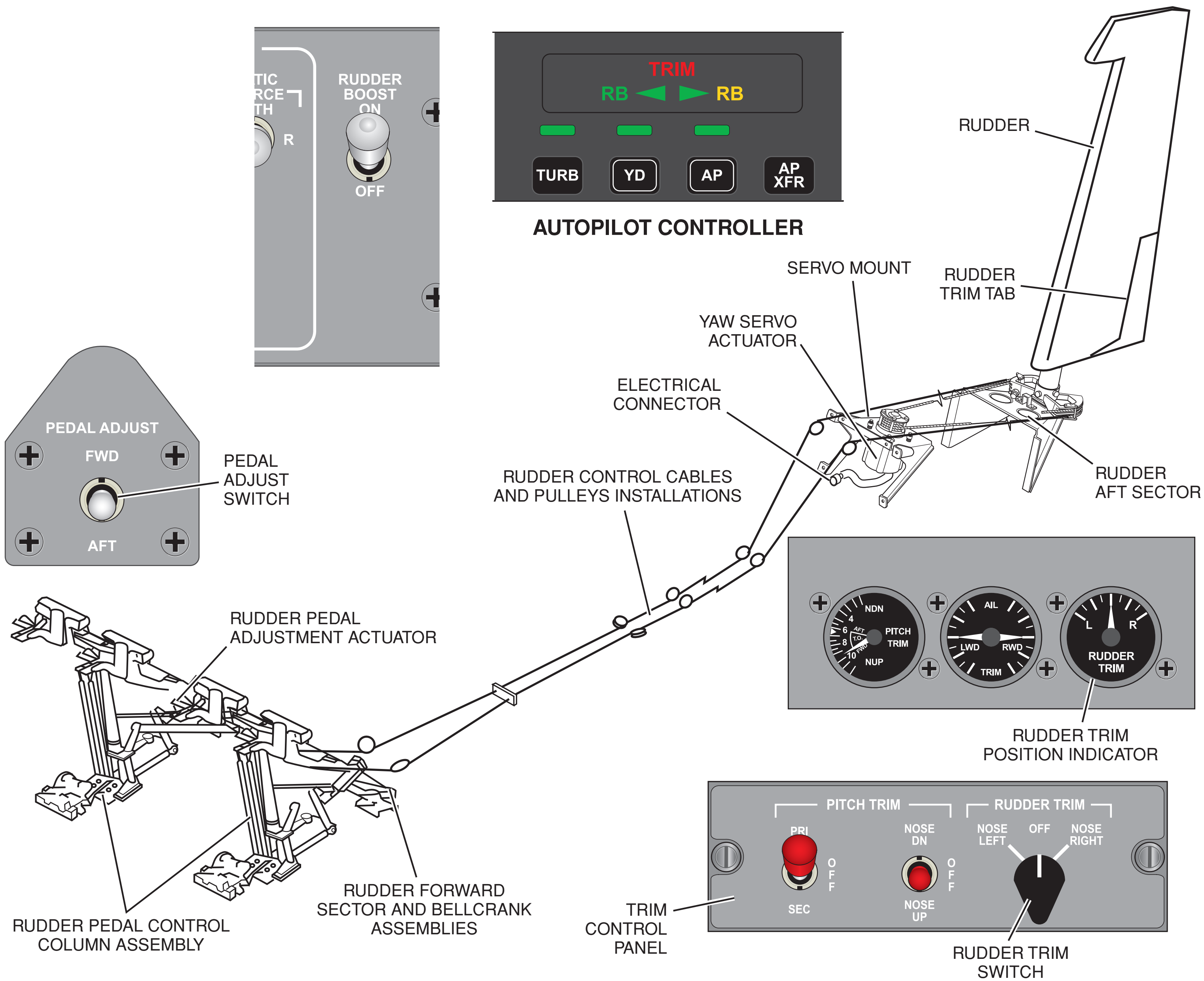


Figure 15-4. Rudder Control System

mounted autopilot controller (Figure 15-4). If a conflict exists between the yaw damper and rudder boost commands, the rudder boost system will override the yaw damper.

Rudder Pedals

Each set of rudder pedals is electrically adjustable with PEDAL ADJUST switches on the pilot and copilot outboard switch panels.

The switches are of the spring-loaded toggle type and have FWD, OFF, and AFT positions. Power (28 VDC) is supplied through the RUDDER PED ADJ circuit breaker on the copilot's circuit-breaker panel.

Rudder Trim Tab

A trim tab, mounted on the bottom of the rudder, is controlled by a trim switch on the center pedestal. Trim position is also indicated on the forward instrument panel.

TRIM SYSTEMS

GENERAL

The ailerons and rudder are trimmed with conventional tabs on the control surfaces as previously described.

The aircraft is trimmed in the pitch axis with the movable horizontal stabilizer. A dual-motor (primary and secondary) actuator moves the leading edge of the horizontal stabilizer up or down in response to pitch trim inputs. Controls and indicators for the trim systems are illustrated in Figure 15-5.

The trim position indicators for pitch, roll, and yaw are all operated on 28 VDC supplied through the TRIM-FLAP-SPOILER indicator circuit breaker on the copilot circuit-breaker panel.

The *AFM* specifies a minimum trim system check that must be accomplished before each flight and an expanded trim system check that must be accomplished once every ten hours of aircraft flight operations.

RUDDER TRIM (YAW)

Control

Rudder (yaw) trim is controlled through the rudder trim switch, spring-loaded to the OFF position (Figure 15-4).

The switch knob is split, providing an upper and a lower half. Both halves must be rotated simultaneously to initiate rudder trim tab motion. This is a safety feature to reduce the possibility of inadvertent trim actuation.

Uncommanded rudder trim can be stopped by depressing and holding either control wheel master switch (MSW).

The rudder trim system operates on 28 VDC supplied through the YAW TRIM circuit breaker on the pilot's circuit-breaker panel. Rudder trim is disabled while either control wheel master switch (MSW) is depressed.

Rudder Trim Indicator

Rudder trim tab position indication is provided by the RUDDER TRIM indicator (Figure 15-4).

AILERON TRIM

Control

Aileron (roll) trim is controlled with either control wheel trim switch located on the outboard horn of each control wheel (Figure 15-5). Each control wheel trim switch is a dual-function (trim and trim arming) switch which controls roll and primary pitch trim. Each switch has four positions; LWD, RWD, NOSE UP, and NOSE DOWN, in addition to the spring-loaded neutral position. The arming button on top of the switch must be depressed and held while simultaneously moving the trim switch in the direction of desired trim action. Actuation of either control wheel trim switch to LWD or RWD (with arming button depressed) will signal the trim tab actuator in the left aileron to move the trim tab as commanded.

Actuation of the pilot trim switch will override actuation of the copilot switch.

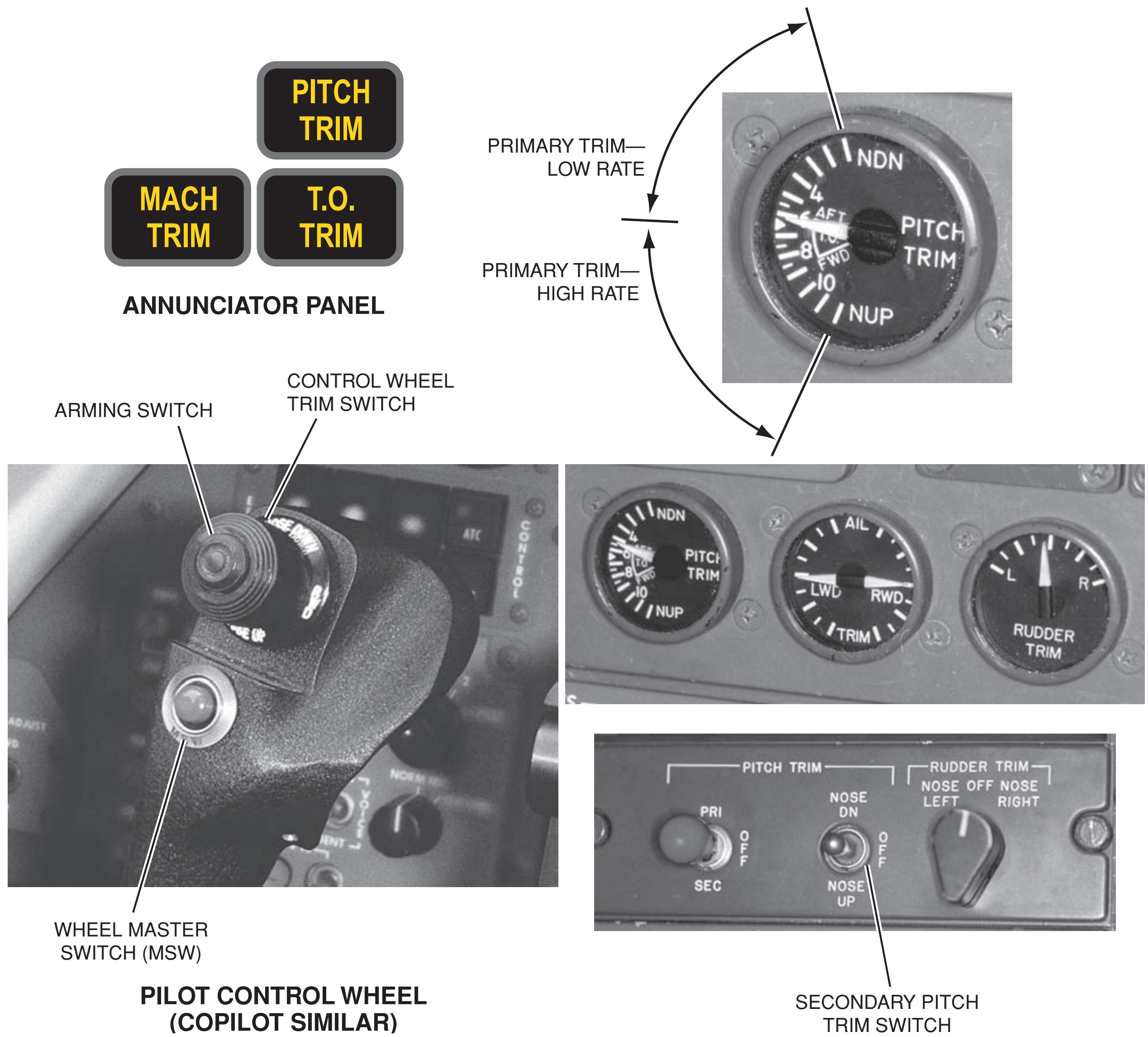


Figure 15-5. Trim Systems Controls and Indicators

Uncommanded aileron trim can be stopped by depressing and holding either control wheel master switch (MSW). Power (28 VDC) for aileron trim is supplied through the ROLL TRIM circuit breaker on the pilot's circuit-breaker panel.

Aileron Trim Indicator

Aileron trim tab position indication is provided by the AIL TRIM indicator (Figure 15-5).

PITCH TRIM

General

Pitch trim is accomplished by repositioning the horizontal stabilizer to the desired trim setting with a dual-motor (primary and secondary) actuator that operates in four modes (Figure 15-6):

Primary Trim Motor

- Primary pitch trim mode
- Mach trim mode (Above 0.70M_I)

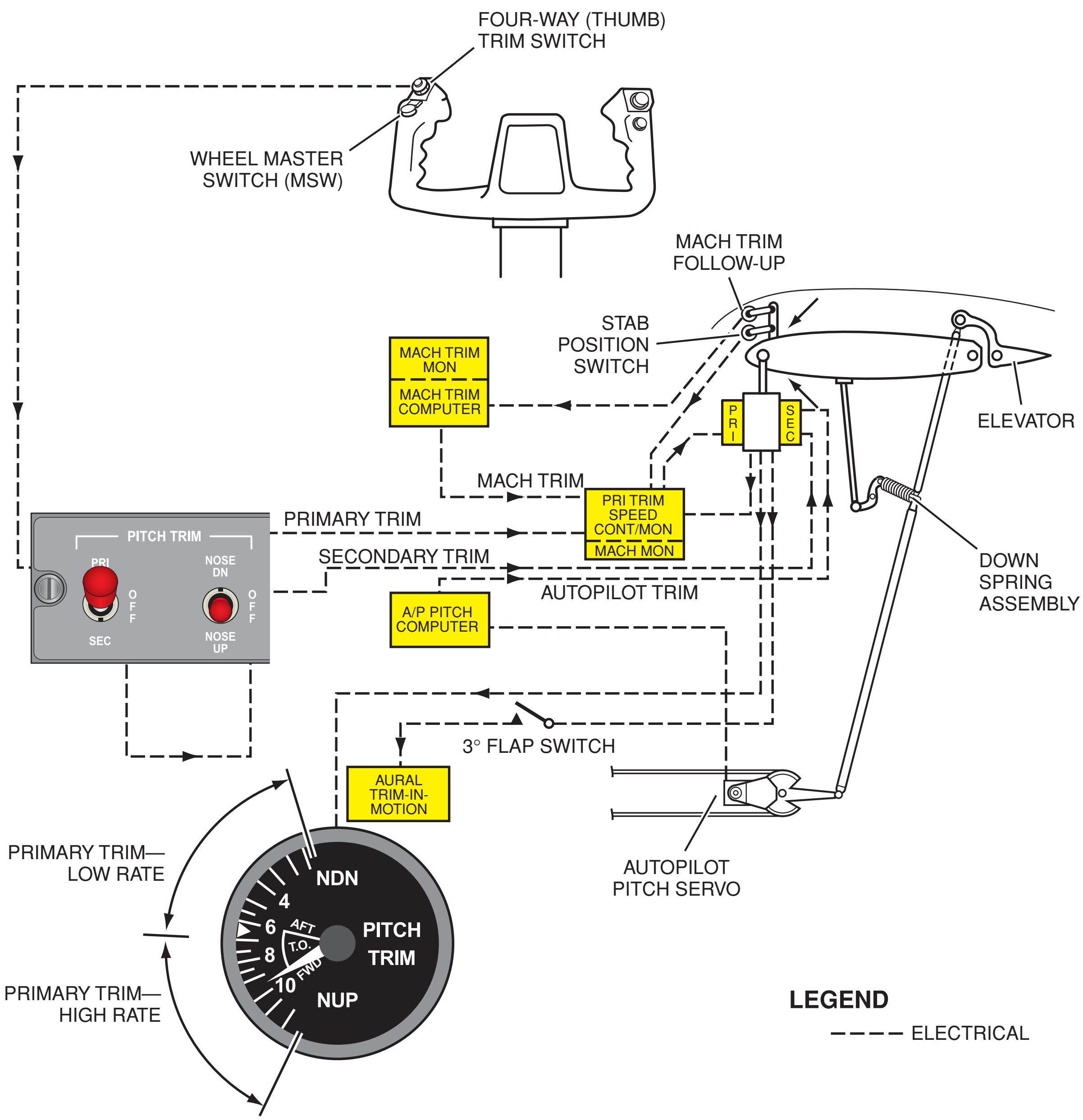


Figure 15-6. Trim System Schematic

Secondary Trim Motor

- Secondary pitch trim mode
- Autopilot pitch trim mode

The pilot-operated primary pitch trim and secondary pitch trim systems are electrically independent systems. Mode selection (primary or secondary) is made with the PITCH TRIM selector switch (see Figure 15-6).

Primary pitch trim is pilot-controlled through either of the control wheel trim switches; secondary pitch trim through the secondary pitch trim toggle switch on the center pedestal (trim control panel).

Mach trim automatically engages at 0.70 M_I if the autopilot is not engaged, using the primary trim motor to adjust pitch trim. Autopilot operation uses the secondary motor to trim in the pitch axis.

Horizontal stabilizer position is displayed on the PITCH TRIM indicator.

Pitch Trim Actuator

The pitch trim actuator is powered by two 28-VDC motors, either of which can move the horizontal stabilizer. Primary trim receives electrical power from the battery-charging bus through a 20-amp current limiter. The control circuit is powered through the PRI PITCH TRIM circuit breaker on the pilot circuit-breaker panel. Primary trim will continue to operate in the EMER BUS mode of operation.

The secondary trim for autopilot operation receives electrical power from the battery charging bus through a power relay. The power relay is closed with power from the right EMER BUS through the SEC PITCH TRIM circuit breaker. Electrical power for pilot-operated secondary trim comes from the right EMER BUS through the SEC PITCH TRIM circuit breaker on the copilot circuit-breaker panel. If the SEC PITCH TRIM circuit breaker is out, the PITCH TRIM annunciator will illuminate. If SEC PITCH TRIM to the autopilot is lost, the red TRIM light will illuminate on the autopilot control panel. Pilot-operated secondary trim is operable in the EMER BUS mode of operation, but autopilot trim is not.

The primary motor operates at two speeds (high or low); the secondary motor operates at a single speed (low). The operating speed of the primary motor is automatically controlled by the primary trim speed controller/monitor, and is based on horizontal stabilizer position.

Primary Trim Speed Controller Monitor

The primary trim speed controller/monitor controls the speed of the primary trim motor during primary trim operation.

Horizontal stabilizer position is measured from 1° (full up) to 12° (full down). The pitch trim indicator has pitch trim markings from 1° to 12° with the full up horizontal stabilizer position annotated as (Aircraft) N DN trim and full down stabilizer position annotated as (Aircraft) N UP trim. The nose of the aircraft will always respond in the opposite direction to the stabilizer position.

The trim speed controller shifts the primary trim speed at approximately 6.5°. This point is marked on the PITCH TRIM indicator with an index mark (►). Primary trim speed above the index (N DN side—high airspeed) will be at a low rate; trim speed below the index (N UP side—low airspeed) will be at a high rate (approximately four times faster than low rate).

The trim speed controller monitors the horizontal stabilizer position (above or below 6.5°) and monitors the actuator speed during primary trim motor operation. If the actuator is operating at a high rate in the low rate range (N DN side of the index), the monitor illuminates the amber PITCH TRIM light on the glareshield annunciator panel (see Figure 15-5). The monitor function can be tested with the SYS TEST switch which will be covered later.

Trim-in-Motion Audio Clicker

A trim-in-motion audible clicker alerts the crew to horizontal stabilizer movement. A series of audible clicks is heard through the headsets and cockpit speakers whenever the pitch trim actuator is operating continuously for more than approximately one second.

The trim-in-motion clicker may or may not sound during Mach trim or autopilot trim, depending on the duration of trim inputs.

The audio trim-in-motion clicker will operate only when the flaps are up.

Power for clicker operation is 28 VDC, supplied through the WARN LTS circuit breaker on the copilot's circuit-breaker panel.

Pitch Trim Selector Switch

The PITCH TRIM selector switch provides primary and secondary mode selection (see Figure 15-5). In the PRI position, primary pitch trim is available using either of the control wheel trim switches. In the OFF position, both trim motors and control circuits are deenergized and the amber PITCH TRIM light illuminates.

In the SEC position, secondary pitch trim is available using the secondary trim switch (see Figure 15-5); this renders primary trim and Mach trim inoperative. The secondary pitch trim switch is spring-loaded to the OFF (center) position.

If pitch trim is attempted with either control wheel trim switch when the trim selector switch is in the SEC position, the PITCH TRIM light will illuminate until the control wheel trim switch is released.

The autopilot can be operated using the secondary trim motor with the trim selector switch in the PRI or SEC position; however, if either control wheel trim switch is actuated with the arming button depressed (see Figure 15-5) or if the secondary trim switch is actuated, the autopilot will disengage.

In the event of inadvertent trim operation (runaway trim) depressing and holding either control wheel master switch (MSW) will inhibit primary and secondary pitch trim. The PITCH TRIM light will illuminate while the control wheel master switch is depressed.

If the control wheel master switch is depressed while trim is activated with the control wheel trim switch (as called for in the preflight trim check), it is necessary to release both switches to reset the system.

The control wheel trim switches were described in this chapter under Aileron Trim.

Pitch Trim Indicator

Horizontal stabilizer trim position indication is provided by the PITCH TRIM indicator (see Figure 15-5). An index, located at 6.5° on the indicator face, marks the primary trim speed changeover point.

A T.O. (takeoff) trim segment is marked from 5.7° to 8.75°. The margins of the T.O. segment are marked AFT and FWD, representing the trim limits for center-of-gravity extremes. Whenever the horizontal stabilizer is not within the range indicated by the T.O. segment, the amber T.O. TRIM light will illuminate (on the ground only).

Pitch Trim Light

The PITCH TRIM light alerts the crew of pitch trim system mismanagement or an abnormality. Six different conditions will cause the light to illuminate. All have been described previously, but are listed again for review.

The first three conditions result from system management; the last three are system malfunctions. The last condition is further described under Mach Trim.

1. Either control wheel master switch (MSW) is depressed.
2. The PITCH TRIM selector switch is in the OFF position.

3. Primary trim is attempted with either control wheel trim switch when the PITCH TRIM selector is in the SEC position.
4. The trim speed monitor has detected the primary trim motor operating at high rate when the horizontal stabilizer is in the low trim rate range.
5. The SEC PITCH TRIM circuit breaker is out.
6. The Mach monitor has detected a Mach trim computer output fault. The MACH TRIM and PITCH TRIM lights will both illuminate in this event.

Trim Speed Monitor

The rotary-type SYS TEST switch is rotated to TRIM OVSP to ground test the trim speed monitor. During the test, the switch applies a false low-trim speed stabilizer position signal to the monitor, causing the amber PITCH TRIM light to illuminate when high-speed trim it initiated. Refer to the *AFM* for the complete trim speed monitor check.

MACH TRIM

General

The Mach trim system is a fully automatic pitch trim system that uses the primary trim motor to increase longitudinal stability and counteract nosedown tendency (Mach tuck) in the transonic region. There is no switch to engage the system; it automatically becomes active at approximately 0.70 M_I if the autopilot is not engaged.

Since the Mach trim system uses the primary motor of the pitch trim actuator, the PITCH TRIM selector switch must be in the PRI position for system operation.

If the autopilot is engaged, it performs Mach trim functions and the Mach trim system is in a passive (standby) mode. In this case, the PITCH TRIM selector switch can be in the PRI or SEC position, since the autopilot uses the secondary trim motor.

The Mach trim system consists of a computer, a Mach trim follow-up on the horizontal stabilizer, and an amber MACH TRIM light on the main annunciator panel. The system operates on 115 VAC supplied through the MACH TRIM circuit breaker on the pilot circuit-breaker panel and 28 VDC supplied to the primary trim system. The Mach trim computer receives airspeed information from the pilot selected ADC 1 or 2 air data computer.

Operation

The air data computer electrically transmits air data (Mach number) to the Mach trim computer and the autopilot pitch computer. At approximately 0.70 M_I, the Mach trim system becomes active (unless the autopilot is engaged). If the aircraft is not manually retrimmed in pitch to compensate for the Mach change, the Mach trim computer will command the appropriate pitch trim change (noseup trim for increased Mach, nosedown for decreased Mach) through the primary motor of the pitch trim actuator (Figure 15-7).

Mach trim is interrupted whenever the airplane is manually trimmed. The system resynchronizes to function about the new horizontal stabilizer position when manual trim is released.

In flight, synchronization may also be accomplished by selecting the MACH TRIM position on the SYS TEST switch and depressing the TEST button.

Monitor Systems

The Mach trim system has two monitors:

- Mach trim monitor
- Mach monitor

If either monitor detects a malfunction in the Mach trim system, the amber MACH TRIM light illuminates and the system disengages. If the Mach monitor has detected a fault, it additionally disables the primary pitch trim system and the amber PITCH TRIM light illuminates.

Also, anytime the Mach trim system is disengaged and Mach is above 0.77 M_I, an overspeed warning horn will sound unless the autopilot is engaged.

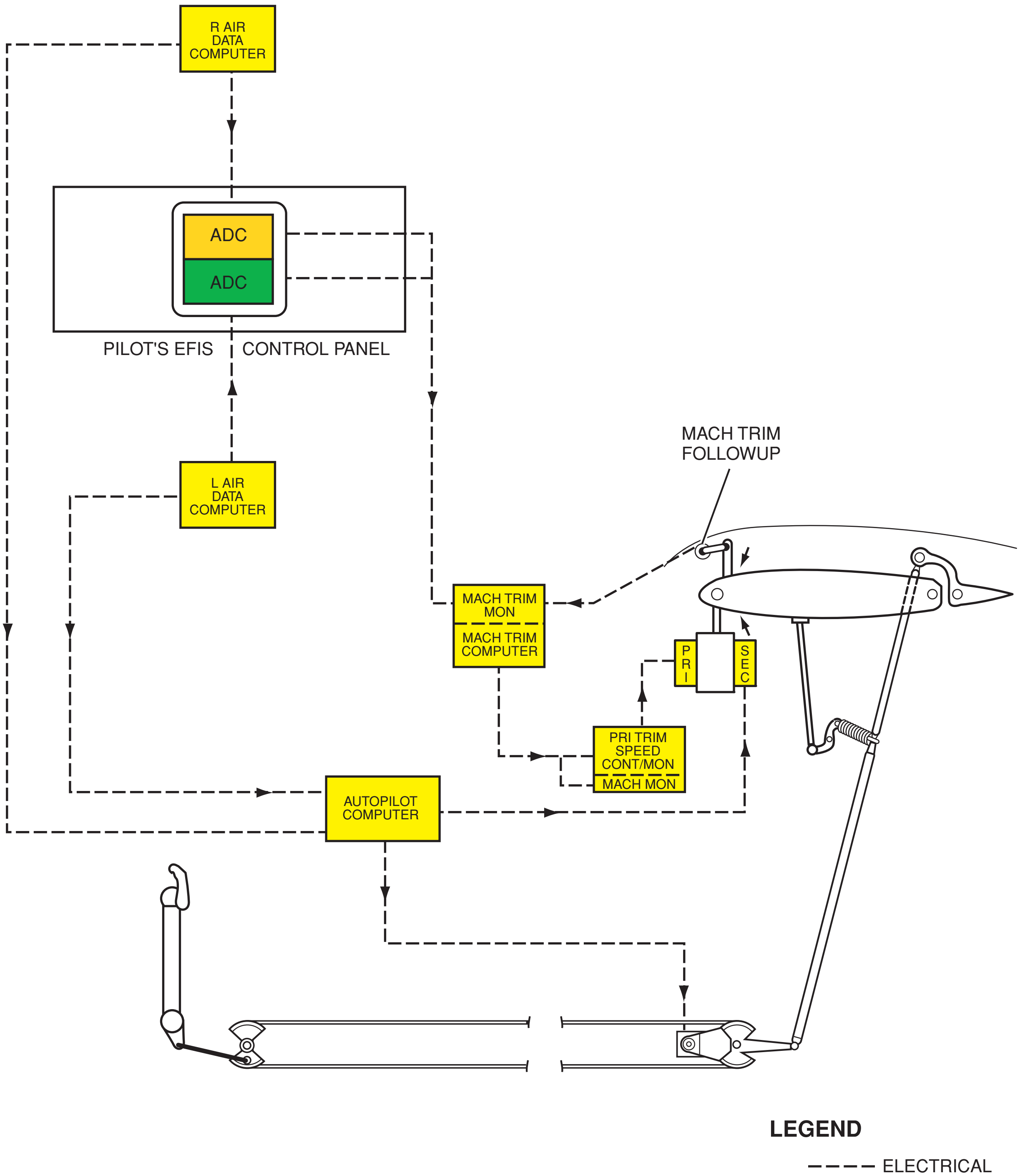


Figure 15-7. Mach Trim System Schematic

Mach Trim Monitor

The Mach trim monitor continuously monitors input signals and power to the Mach trim computer, and compares signal inputs from the air data computer (Mach) and the Mach trim follow up on the horizontal stabilizer.

If the Mach trim monitor does not receive a corresponding signal change from the Mach trim follow up when the air data computer signal changes (Mach change), a malfunction is indicated. In the event of power loss to the Mach trim computer, loss of input signals, or a Mach number/horizontal stabilizer trim position error, the Mach trim monitor will disengage Mach trim and turn on the MACH TRIM light. If speed is above 0.77 M_I and the autopilot is not engaged, the overspeed warning horn will sound. If the fault clears or power is restored, the system can be resynchronized by selecting MACH TRIM position on the SYS TEST switch and depressing the TEST button.

Mach Monitor

The Mach monitor function is provided by circuitry within the primary trim speed controller/monitor. If the Mach trim signals exceed normal limits, the Mach monitor will disengage Mach trim, disable primary pitch trim, and illuminate the MACH TRIM and PITCH TRIM lights. If speed is above 0.77 M_I and the autopilot is not engaged, the overspeed warning horn will sound. If the fault clears or Mach trim circuitry is isolated by pulling the MACH TRIM circuit breaker, primary pitch trim can be reinstated by cycling the PITCH TRIM selector switch to SEC PITCH TRIM and back to primary.

Mach Trim System Tests

The rotary SYS TEST switch is used to test Mach trim system operation and both Mach trim system monitors while the airplane is on the ground. In flight, the MACH TRIM position on the SYS TEST switch may be used to resynchronize the system, if necessary.

When the aircraft is on the ground, the MACH TRIM position of the SYS TEST switch is selected and the TEST button depressed to initiate the Mach trim and Mach trim monitor test. The test switch inserts a signal (Mach trim) that causes the horizontal stabilizer to move in the N UP trim direction. Since there is no corresponding airspeed change, the Mach trim monitor senses a Mach/horizontal stabilizer position error. Mach trim disengages and the amber MACH TRIM and PITCH TRIM lights illuminate. The system test also induces a Mach trim computer fault signal to the Mach monitor which causes the PITCH TRIM annunciator to illuminate if the Mach monitor is functioning. The overspeed warning horn sounds when Mach trim disengages during the test. Figure 15-8 illustrates the test. Refer to the *AFM* for the complete Mach trim check.

SECONDARY FLIGHT CONTROLS

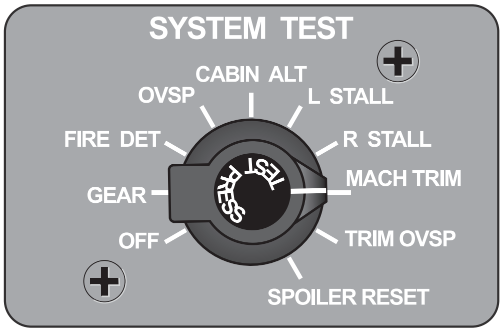
FLAPS

General

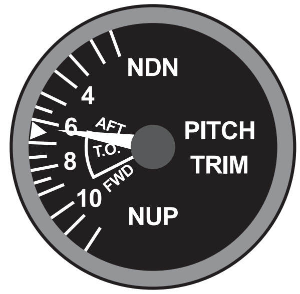
The fowler-type, single-slotted flaps are electrically controlled and hydraulically actuated. Flap motion is a combination of aft movement to increase lift and downward tilting to increase drag. Left and right flaps are cable interconnected to preclude asymmetrical operation.

A position switch is mechanically connected to each flap. Switch action occurs at 3°, 13°, and 25° of flap travel, providing flap position information to the landing gear warning, stall warning angle of attack computer, spoiler warning, spoileron, yaw damper, trim-in-motion audio clicker, autopilot, and flight director systems. Separate flap limit switches (7°/9° and 19°/21°) automatically maintain flap position at the selected setting when selected to flaps 8° or flaps 20°.

The flap control system operates on 28 VDC, supplied through the FLAPS circuit breaker on the copilot circuit-breaker panel. The flaps cannot be operated from the emergency batteries; however, the flaps are powered through the right

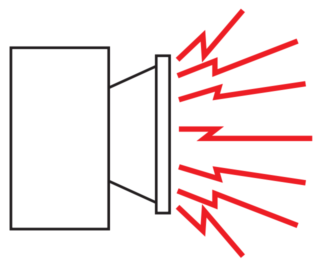


1. SELECT MACH TRIM AND DEPRESS TEST BUTTON.



2. OBSERVE NOSEUP TRIM 1 TO 3 SECONDS.

3. MACH TRIM AND PITCH TRIM ANNUNCIATORS ILLUMINATE WHEN MACH TRIM STOPS (DISENGAGES).



4. OVERSPEED WARNING HORN SOUNDS.

Figure 15-8. Mach Trim Test

emergency bus and can be operated with the emergency bus selected as long as there is energy remaining in the aircraft main batteries.

Flap Selector Switch

The flap selector switch has four positions: UP, 8°, 20°, and DN (40°) with detents at the 8° and 20° positions (Figure 15-9). When the flap selector is raised from the 20° or DN position, a more positive stop is met at the 8° detent. The selector must be pulled out of the 8° detent to raise the selector to UP.

Flap Position Indicator

A vertical-scale FLAP position indicator is mounted on the center instrument panel (Figure 15-9). The flap indicator has the flap limit speeds (V_{FE}) placarded on the left side of the indicator.

Left flap position is electrically transmitted to the indicator. The indicator operates on 28 VDC, supplied through the TRIM-FLAP-SPOILER circuit breaker in the TRIM-FLT CONT group of circuit breakers on the copilot's circuit-breaker panel. The indicator will indicate DN with loss of electrical power, regardless of actual flaps position.

Operation

When the flap selector switch is placed in the DN position, the flaps extend to 40° and are held down by hydraulic pressure (Figure 15-9). If airloads become excessive due to increased airspeed, a relief valve in the flaps down line will open, allowing partial retraction of the flaps.

If transient airloads cause the flaps to move, they will automatically return to the selected position.

When UP is selected, the flaps retract to 0°.

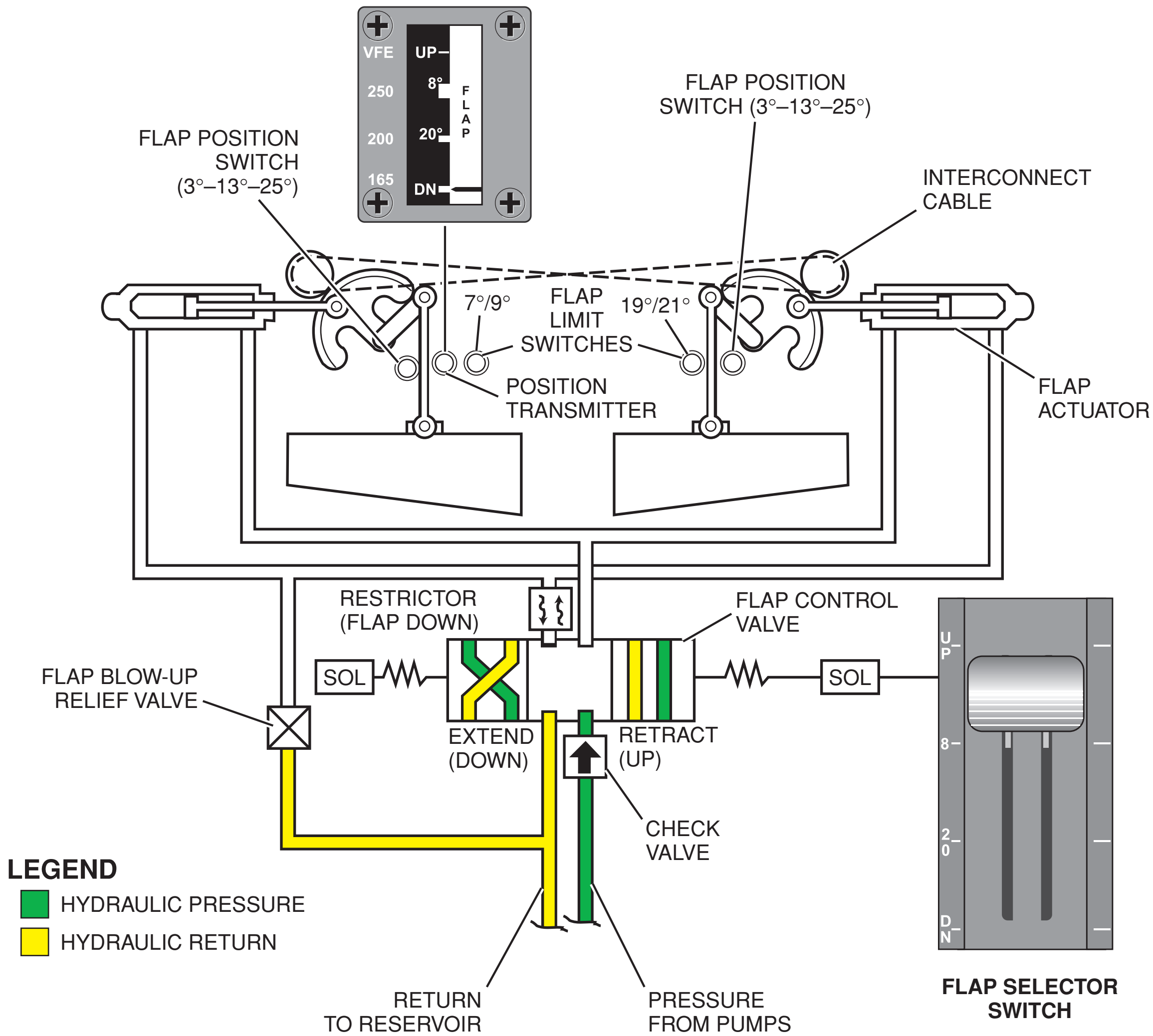


Figure 15-9. Flap System

SPOILERS

The spoiler, located on the upper surface of the wings forward of the flaps, may be extended symmetrically for use as spoilers (spoiler mode) or asymmetrically for aileron augmentation when the flaps are extended (spoileron mode).

The spoilers may be extended symmetrically by placing the lever to EXT or they may be extended automatically by placing the lever in the ARM position. The spoilers may be extended fully or to intermediate position in flight. The autospoiler mode is selected by placing the spoiler switch to ARM prior to takeoff or landing. When the thrust levers are moved to IDLE and the aircraft is on the ground, the spoilers automatically extend (weight on wheels auto spoiler mode).

The spoilers are hydraulically actuated and electrically controlled either by the SPOILER lever (spoiler mode) or by the spoileron computer (spoileron mode).

Both modes require 28-VDC and 115-VAC electrical power, supplied through the SPOILER and SPOILERON circuit breakers, respectively, on the copilot circuit-breakers panel.

If either circuit breaker is pulled or either power source is lost in flight, the spoilers will retract (if extended) and will be inoperative in both modes. Spoiler mode operation does not require 115-VAC power on the ground.

A pressure relief valve in the system allows the spoilers to partially blow down at excessive airspeeds. In the event of main system hydraulic failure, the spoilers will blow closed and be inoperative. Spoilers cannot be operated with hydraulic pressure from the auxiliary hydraulic pump.

The spoiler modes, when selected, will override the spoileron mode if it is operating.

When airborne, flaps and spoilers should not be extended simultaneously (SPOILER EXT light will flash). To do so could cause structural damage to the flaps.

Operation (Spoiler Mode)

The spoilers can be symmetrically extended or retracted with the SPOILER LEVER (Figure 15-10) located on the top left hand side of the forward pedestal assembly, adjacent to the thrust levers. There is a RET position (full forward) and an EXT position (full aft) and detents at ARM and two other intermediate positions. A vertical-scale spoiler position indicator for RET (0°), 10°, 20°, 30°, and 40° is mounted on the center instrument panel.

The EXT spoiler lever position commands full up extension depending upon airspeed; second partial position commands approximately 20° up, and first partial position commands approximately 10° up. When the spoiler lever is placed to any position aft of ARM with the aircraft on the ground, spoilers will fully extend. The white SPOILER EXT annunciator illuminates steady when either spoiler rises above 1° (except during aileron augmentation—spoileron mode).

The SPOILER EXT annunciator flashes if flaps are extended beyond 3° during spoiler operation. RET position retracts the spoilers and extinguishes the SPOILER EXT annunciator.

Since spoiler deployment reduces lift, nose-down pitch should be anticipated and offset with control pressure and pitch trim.

When RET is selected, the SPOILER EXT light goes out and the spoilers are locked down by internal locks within the actuators. The resultant noseup pitch motion can be countered with control pressure and trim.

Operation (Autospoiler)

The autospoiler system is pilot controlled through the spoiler lever on the forward pedestal. The autospoiler system includes a green SPOILER ARM light which illuminates when the spoiler lever is positioned to ARM. Failure of the SPOILER ARM light to illuminate indicates a malfunction.

The spoilers will automatically extend and the white SPOILER EXT light will illuminate when the following conditions exist (Figure 15-11):

- The spoiler system is armed.
- Both squat switches are in the ground mode.
- Both thrust levers are at IDLE or CUT-OFF.

On aircraft SN 60-079, 081, 094 and subsequent, and on aircraft incorporating SB 60-27-6 (wheel spin up kit), the spoilers will automatically extend and the white SPOILER EXT light will illuminate when the following conditions exist:

- The spoiler system is armed.
- The antiskid system has detected wheel spin up.
- Both thrust levers are at IDLE or CUT-OFF.

When the spoilers have been automatically deployed, advancing either thrust lever above IDLE will cause the spoilers to retract.

Autospoilers are only operational when ARM is selected on the spoiler lever. The autospoilers system is used to automatically extend the spoilers in the event of an aborted

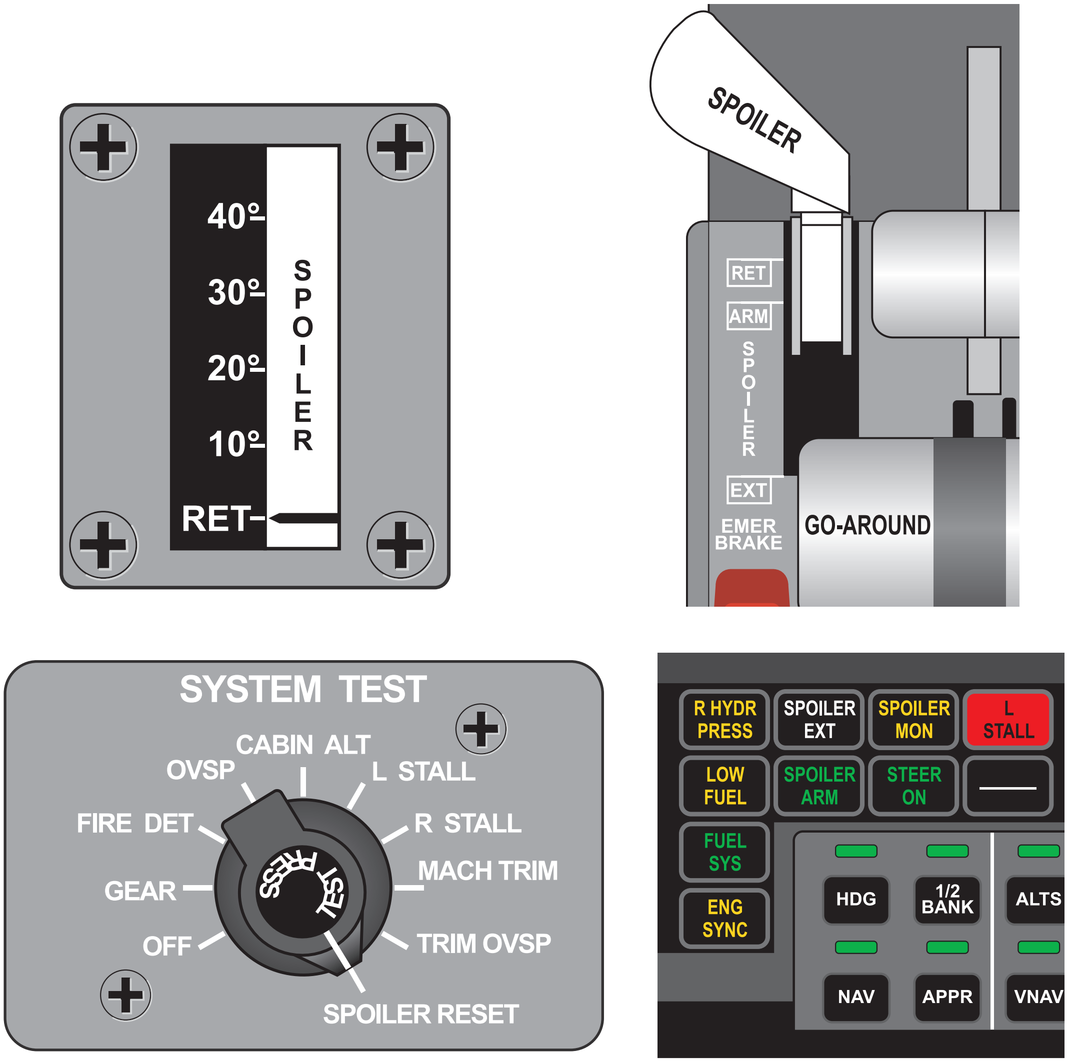


Figure 15-10. Spoiler Controls and Indicators

takeoff. However, to standardize procedures for aborted takeoff, move the SPOILER switch to EXT even with autospoilers armed.

Autospoilers may also be used for landing, but because of possible delayed spoiler deployment during soft landing, EXT should be selected even if the autospoilers are armed.

Operation (Spoileron Mode)

During spoileron (aileron augmentation) mode of operation, the spoilers are independently extended and retracted on a one-to-one ratio with the upgoing aileron to increase lateral control in the landing configuration. Aileron augmentation (spoilerons) increase roll control authority (Figure 15-12).

The spoileron mode is automatically engaged when the flaps are lowered beyond 25° and the SPOILER lever is in the RET or ARM position.

The spoileron computer continuously monitors aileron position. When the ailerons are displaced from neutral with the manual controls, the computer commands extension of the spoiler on the wing with the raised aileron. The spoiler on the opposite wing remains retracted. Spoiler extension is limited to approximately 15° during spoileron operation.

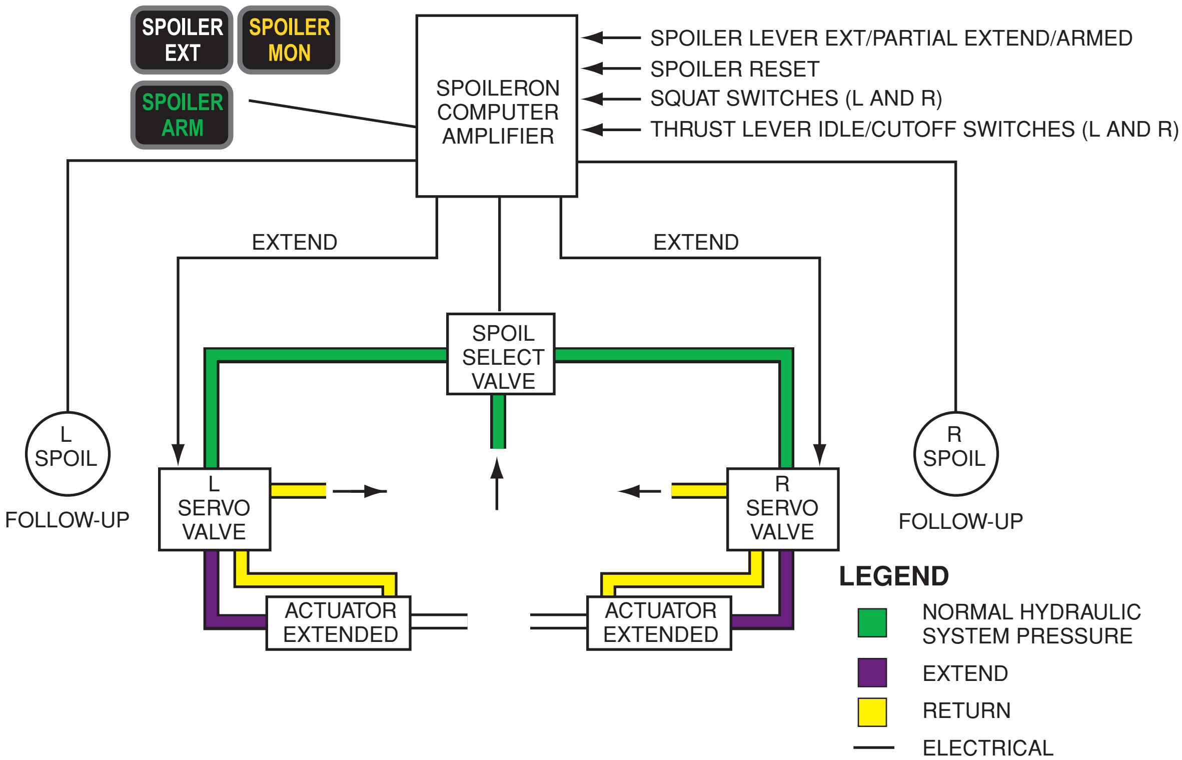


Figure 15-11. Spoiler/Autospoiler Operation

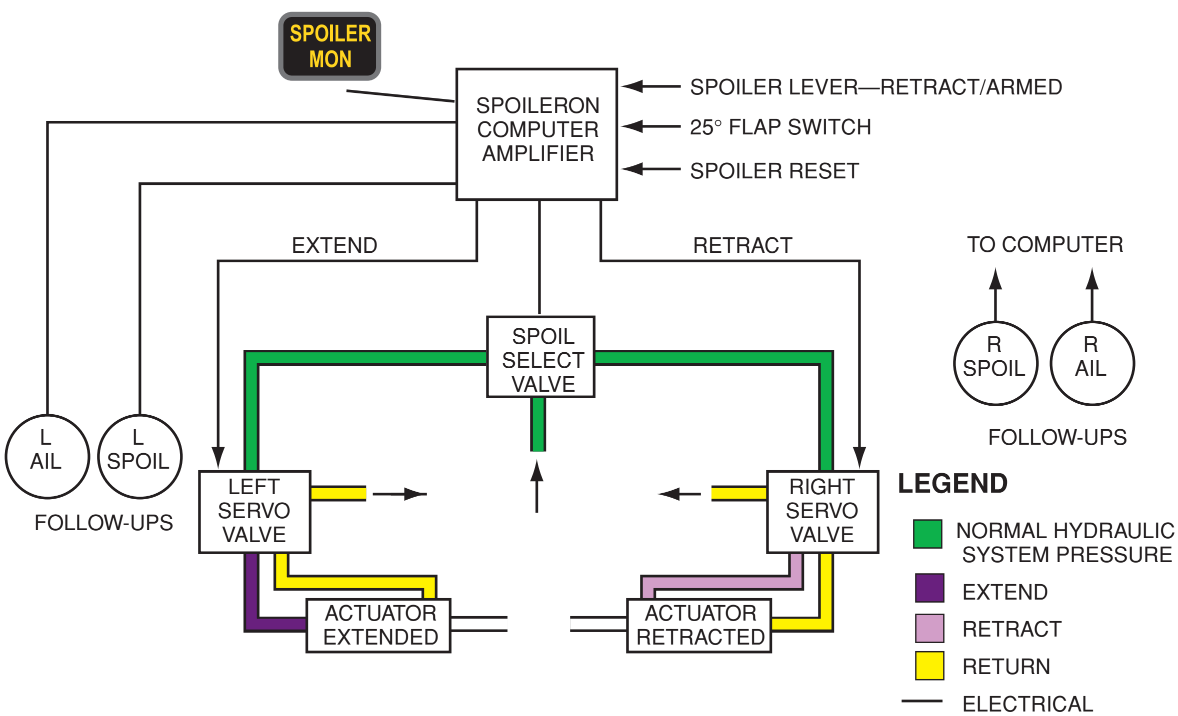


Figure 15-12. Spoileron Operation (Left Aileron Up)

Self-Monitor System

The computer monitors spoiler and spoileron modes of operation. The SPOILER MON (spoiler monitor) annunciator illuminates whenever monitor circuits in the spoileron computer detect a malfunction during spoiler, autospoiler, or spoileron mode, whenever unequal spoiler extension (6° or more difference) occurs during the spoiler/autospoiler mode, or whenever the spoiler and aileron deflection is unequal (6° or more difference) in the spoileron mode. The SPOILER MON annunciator also illuminates if electrical power to the computer is lost through the DC SPOILER or AC SPOILERON circuit breakers. If a malfunction or monitored split occurs, the amber SPOILER MON light will come on and all modes will be inoperative in flight. However, spoiler mode may still be operative for ground operation.

Spoiler Reset Test Switch

SPOILER RESET is located on the rotary SYSTEM TEST panel (see Figure 15-10). If a malfunction occurs in any mode (SPOILER MON light on), momentarily depressing the test button will restore spoiler/spoileron operation, provided the malfunction has cleared. If the SPOILER MON light does not extinguish, both modes will be inoperative in flight.

The SPOILER RESET/TEST switch is used during the spoileron/spoiler preflight check of monitor circuit operation. Holding the switch depressed in RESET inhibits spoileron defection. Therefore, if the control wheel is turned while holding the switch in RESET, the SPOILER MON light should illuminate after the aileron has deflected approximately 6°. The system can be reset by releasing the SPOILER RESET switch and centering the control yoke and momentarily depressing RESET. The SPOILER RESET switch is also used to check slamdown of the autospoiler from an extended position in the armed mode. Refer to the *AFM* for the complete spoileron/spoiler check.

YAW DAMPER

GENERAL

The yaw damper provides automatic stabilization about the yaw axis.

The system can be used with or without the autopilot engaged. The system will provide full-time yaw damping by applying rudder against transient motion in the yaw axis. Delta fins on the tail section add stability to the aircraft, reducing the need for yaw dampers. Only one yaw damper is installed, and the *AFM* allows the aircraft to be dispatched without an operational yaw damper (see Figure 15-4).

The yaw damper uses the autopilot/flight guidance computer yaw output to drive the yaw servo. The autopilot must be operative for yaw damper operation but does not have to be engaged. Also the yaw damper receives yaw rate and lateral acceleration from the attitude heading system (AHS). Both AHSs must be working for the autopilot/yaw damper to operate.

The pilot can override the yaw damper at any time, if required. When the flaps are lowered to 3° or more, yaw damper authority (force) is reduced so that landings may be accomplished with the yaw damper engaged. When the auto-pilot is engaged, the yaw damper has full authority regardless of flap position.

The yaw damper will normally be engaged throughout the flight except for takeoff or while trimming rudder. In the event a no-flap landing is required, the yaw damper must also be disengaged prior to landing.

The yaw damper servo uses 28 VDC, supplied through the ROLL-YAW SERVO circuit breaker in the AFCS group of circuit breakers on the pilot circuit-breaker panel.

The yaw damper is powered through the autopilot. A self-test is completed within the autopilot when DC electrical power is initially applied to the airplane electrical system. Autopilot power is 28 VDC through the AP-1 circuit breaker on the pilot circuit-breaker panel or AP-2 circuit breaker on the copilot circuit-breaker panel.

OPERATION

Rudder pedal force sensors on each crew member's rudder pedal assembly provide force data to a yaw force interface box which outputs signals to the flight control computers (FCCs), a component of the integrated avionics processing system (IAPS). The FCCs command rudder boost and also drive the rudder servo actuator for yaw dampening and controlling the aircraft laterally in certain autopilot modes.

The yaw damper is powered whenever the airplane electrical system is powered. The autopilot need not be engaged for the yaw damper to work. The yaw damper engages whenever the autopilot is engaged, or may be engaged separately by depressing the YD button on the autopilot controller (see Figure 15-4). A green light above the YD button illuminates when damper is engaged. The yaw damper can be subsequently disengaged by momentarily depressing the YD button if so desired.

Also, the yaw damper can be disengaged by momentarily depressing the wheel master switch. Whenever the yaw damper disengages, the yaw damper/autopilot disconnect tone sounds and produces a flashing yellow YD annunciator on the pilot and copilot primary flight display (PFD). Intentional disengagement (yaw damper or control wheel master switches were pressed) causes the flashing annunciator to cancel after five seconds.

If, however, disengagement was automatic due to a system failure, the flashing persists for ten seconds and is then replaced by a steady yellow YD message on both PFDs. The yellow YD can be cancelled by momentarily depressing either control wheel master switch (MSW).

RUDDER BOOST

For the rudder boost system to activate the following four conditions must exist:

- Both avionics master switches ON.
- Flaps extended greater than 3°.
- Rudder boost switch ON.

The rudder boost system responds to total crew rudder pedal force in excess of 50 pounds by applying torque to the rudder capstan proportionate to the rudder pedal pressure being applied.

The RUDDER BOOST switch, located on the pilot switch panel, arms the system (see Figure 15-4). When the switch is ON, rudder boost is available and will be engaged by the FCC's if required. The pilot may elect to have the rudder boost ON or OFF for takeoff. Takeoff performance is provided for both conditions in the Performance section of the *AFM*.

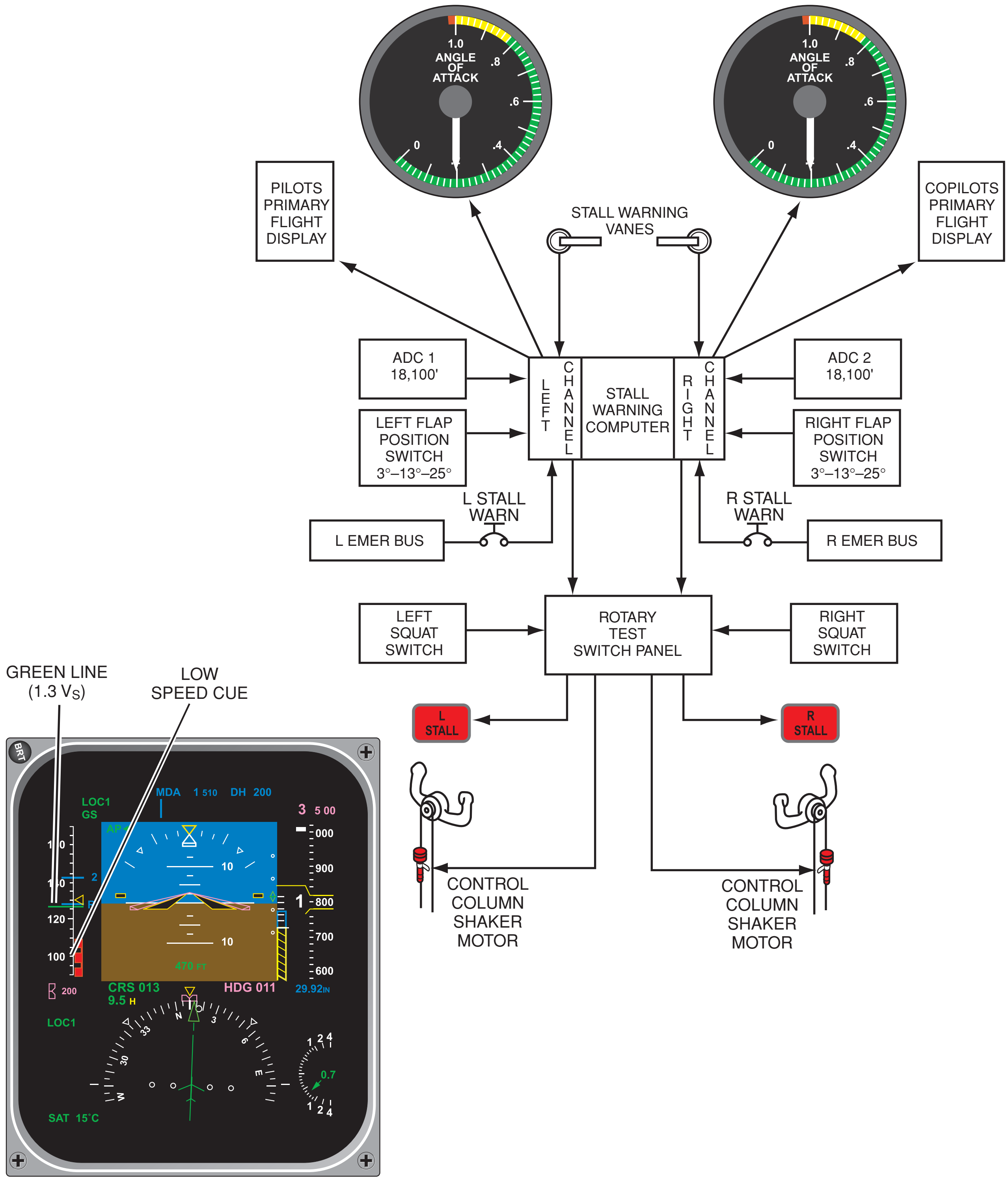
In the event of a malfunction in the rudder boost system, either control wheel master switch (MSW) can be depressed and held to disconnect the rudder boost (yaw) servo.

Annunciation is provided by amber and green RB lamps located on the autopilot control panel (see Figure 15-4). An amber RB denotes rudder boost is not available due to a malfunction, an absence of power (circuit breaker pulled) or disengagement via the rudder boost or control wheel master switches. A green RB indicates the system is actively applying torque to the rudder. The absence of both amber and green indicators implies the system is healthy and available, but rudder pedal forces and/or aircraft conditions do not warrant rudder boost.

STALL WARNING SYSTEM

GENERAL

There are no stall warning switches in the cockpit. It is not necessary for the stall warning system in these aircraft to be as extensive due mainly to the addition of the delta fins. At high angles of attack, the delta fins provide a nosedown moment that prevents the aircraft from entering the deep stall that T-tailed aircraft are most susceptible to. The boundary layer energizers (BLEs) on the leading edge of the wings improve slow speed handling characteristics.



PFD AIRSPEED DISPLAY

Figure 15-13. Stall Warning System

The dual stall warning system provides visual and tactile warning of an impending stall. The system contains the following components: left and right angle of attack (AOA) vanes, a dual channel (left and right) computer, L and R STALL warning lights, and two stick shaker motors (Figure 15-13).

Power is applied to the dual stall warning systems any time the circuit breakers are in and electrical power is applied to the aircraft. Operation of the stall warning lights and the shaker motors is inhibited on the ground except when the left or right stall warning test is activated. The systems become fully operational approximately seven seconds after the squat switch relay goes to the airborne mode. The dual angle of attack indicators are operational on the ground and in flight.

Flap position switches and altitude switches (one for each channel) provide bias information to the computers. Angle of attack indicators are adjusted (biased) by the computer as the flaps are extended past the 3°, 13°, and 25° positions. The altitude switches signal the computer at approximately 18,100 feet, and AOA indications are shifted upward to compensate for higher indicated stall speeds at high altitude. The left and right systems are independent in that they use separate electronics, stall vanes, altitude switches, shaker motors, and flap switches. The systems operate on 28 VDC supplied through the L and R STALL WARN circuit breakers in the TRIM–FLT CONT group of circuit breakers on the left and right circuit-breaker panels. The stall warning systems continue to operate if the emergency bus mode of operation is selected.

STALL WARNING INDICATORS

The system presents indications of approaching stall (and stall) with angle of attack indicators, warning lights, and the use of control column stick shakers (Figure 15-13).

Angle of Attack Indicators (AOA)

The AOA indicators translate signals from the computer into visual indications of stall margin. The face of the indicators is divided into three color segments—green, yellow, and red. Green is the normal operating range. The yellow segment represents caution (approaching stall). The shaker will activate when the AOA pointer enters this segment. The green-yellow margin represents approximately 10% above stall. The red segment represents danger—the angle of attack is just below aerodynamic stall. The low speed cue on the pilot's flight display (PFD) also receives data from the stall warning (explained in Chapter 16).

Warning Lights

The L and R STALL warning lights will come on and flash when the respective AOA indication pointers enter the yellow segment (shaker range). The L and R STALL warning lights will illuminate steady in the red segment. Steady illumination of the lights at any other time indicates a computer power loss or a circuitry malfunction.

Stick Shaker

Stick shaker motors are attached to the front side of each control column. Actuation of the shakers causes a high-frequency vibration in the control columns. In addition, the AOA computers provide inputs that display 1.3 V_S (greenline) and shaker speed (low speed cue) on the PFD airspeed displays (Figure 15-13).

OPERATION

During flight, the AOA vanes align with the local airstream. Vane-operated transducers produce a voltage proportional to airplane angle of attack. These transducer signals, along with information from the flap position switches and altitude switches, are sent to the appropriate computer channel. If angle of attack is increased to a point which corresponds to a speed of 10% above stall speed, the AOA indicator pointers will enter the yellow segment, the L and R STALL lights will flash, and the stick shakers will actuate. If the angle of attack is increased to a point just above stall speed, the AOA pointers will enter the red segment, and the L and R STALL lights will illuminate steadily.

Stall Warning System Test

The test is initiated by rotating the SYS TEST switch to L or R STALL (as applicable) and then depressing and holding the TEST button. The corresponding AOA indicator pointer will begin to sweep from the green segment toward the red segment. As the pointer passes the green-yellow margin, the applicable (L or R) STALL light will begin to flash and the applicable shaker will actuate. The flaps can be up or down during the stall warning test. If the AOA indicators are observed during flap extension or retraction before flight, the flap bias function can be checked.

OVERSPEED WARNING

GENERAL

The overspeed warning system provides audible overspeed warning in the event airplane speed exceeds V_{MO} or M_{MO}. A visual indication of overspeed is also provided by the overspeed cue on the PFD airspeed displays (explained in Chapter 16).

OPERATION

The overspeed warning horn is functional whenever the aircraft electrical system is powered and circuit breakers are engaged and overspeed is exceeded.

The following is a summary of the conditions under which the overspeed warning horn will actuate.

1. Airspeed exceeds 300 KIAS up to 8,000 feet.
2. Airspeed exceeds 340 KIAS up to 20,000 feet. Decreasing from 340 KIAS to 330 KIAs by 23,000 feet and 330 KIAS to 26,700 feet.
3. Mach exceeds an airspeed corresponding to 0.81 M_I up to 37,000 feet.
4. Mach exceeds airspeed corresponding to 0.81 M_I at 37,000 feet. Decreasing to 0.78 M_I by 43,000 feet.
5. Mach exceeds an airspeed corresponding to 0.78 M_I above 43,000 feet.
6. Mach exceeds an airspeed corresponding to 0.77 M_I with the Mach trim inoperative and the autopilot not operating.

Overspeed Warning Test

The overspeed warning test is the position that's labeled OVSP on the rotating system test selector knob. With the OVSP position selected, depressing and holding the TEST button will cause the overspeed warning horn to sound briefly and cease, again sound briefly and cease, then sound continuously until the TEST button is released. Refer to the *AFM* for the complete overspeed warning check.

See Chapter 16 for a description of the autopilot/flight director overspeed modes.

QUESTIONS

1. Which of the following systems use(s) the pitch servo to position the elevators?

A. Autopilot and Mach trim
B. Autopilot and shaker
C. Stick shaker and Mach trim
D. Autopilot only

2. If the pitch servo makes an uncommanded elevator input, the pilot can disengage the pitch servo clutch by:

A. Momentarily depressing the synch switch
B. Depressing and holding the control wheel master switch
C. Moving the PITCH TRIM selector switch to OFF
D. Engaging the autopilot

3. The airplane is trimmed in the pitch axis by:

A. Movable trim tabs on the elevators
B. Canards
C. The movable horizontal stabilizer
D. A down-spring on the elevators

4. To enable pitch trim through the control wheel trim switches, the PITCH TRIM selector switch must be in the ____________ position(s).

A. PRI or SEC
B. PRI, OFF, or SEC
C. PRI
D. SEC

5. Illumination of the amber MACH TRIM light indicates:

A. Mach trim is not operating
B. The secondary trim motor is inoperative
C. The autopilot is engaged above 0.70 M_I
D. The trim speed controller/monitor has detected a trim speed error

6. The___________ and ______________ systems are operational with the PITCH TRIM selector switch in the SEC position.

A. Primary pitch trim and Mach trim
B. Secondary pitch trim and Mach trim
C. Secondary pitch trim and primary pitch trim
D. Secondary pitch trim and autopilot pitch trim

7. Illumination of the amber PITCH TRIM light in flight could indicate:

A. PITCH TRIM selector switch is off or the control wheel master switch is depressed
B. Attempting to trim with the control wheel trim switch with the PITCH TRIM selector in SEC
C. Trim speed controller/monitor has detected primary trim speed error
D. All the above are true

8. Illumination of the MACH TRIM and PITCH TRIM lights at the same time indicates:

A. Primary trim speed error
B. Primary trim has failed
C. Mach monitor has detected a Mach trim computer signal error
D. Either B or C

9. The overspeed warning horn sounding at 0.77 M_I indicates:

A. Airspeed has reached M_{MO} below 8,000 feet
B. The autopilot is not engaged, and Mach trim is not operating.
C. The autopilot is engaged, but Mach trim has failed
D. The stick puller is inoperative

10. In the event of runaway trim, both pitch trim motors can be disabled by:

A. Depressing and holding either control wheel master switch
B. Moving the PITCH TRIM selector switch to OFF
C. Moving the PITCH TRIM selector switch to EMER position
D. A or B

11. The two-speed primary trim motor operates at low rate when:

A. The flaps are up.
B. The flaps are down.
C. The horizontal stabilizer is on the N DN side of the index ( ) on the pitch trim indicator.
D. The horizontal stabilizer is on the N UP side of the index ( ) on the pitch trim indicator.

12. The OVSP position on the rotary SYS TEST selector switch is used to test:

A. Mach trim and Mach trim monitor
B. Overspeed warning horn and stick shaker
C. Overspeed warning horn, shaker, and ADC
D. Overspeed warning system

13. In the event of airplane electrical failure, the flap position indicator will:

A. Be powered by the EMER BAT and indicate actual position of the flaps
B. Not be powered and will freeze at last flap position
C. Not be powered and will go to full-scale down defection regardless of flap position
D. None of the above

14. With the spoileron system inoperative:

A. The airplane must not be flown.
B. The airplane can be flown, without restrictions.
C. The airplane can be flown, but the SPOILERON circuit breaker should be pulled and altitude limited to 38,000 feet.
D. None of the above

15. A flashing white SPOILER light indicates:

A. Spoilers are split more than 6°.
B. Spoiler-aileron relationship has exceeded 6°.
C. Spoiler system is inoperative
D. Spoilers are extended and flaps are extended past 3°.

16. The SPOILER RESET/TEST switch is used to:

A. Retract the spoilers in the event of a malfunction.
B. Induce a fault during the spoileron monitor test.
C. Reset the spoiler/spoileron system when the SPOILER MON light illuminates.
D. B and C

17. If the yaw damper is inoperative:

A. The airplane may be flown, but altitude is restricted to 20,000 feet.
B. The airplane may be flown, but altitude is restricted to 38,000 feet.
C. The airplane may be flown.
D. The airplane must not be flown.

18. Yaw damper authority is reduced, making it easier for the pilot to overpower the yaw damper when:

A. The gear is lowered.
B. Below 200 KIAS
C. Flaps are up.
D. Flaps are set to 8° or lower.

19. When the Angle of Attack indicator pointers enter the yellow segment:

A. The stall warning horn sounds
B. STALL WARN lights illuminate steady
C. The shakers activate, and the STALL WARN lights flash
D. The shakers activate and the stall warning horn sounds

20. The electrical power source for the stall warning system is the:

A. Battery buses
B. Battery-charging bus
C. Main DC buses
D. Emergency buses

21. If either L or R stall warning system is inoperative:

A. The airplane can be flown, provided the STALL WARN circuit breaker is pulled for the inoperative system.
B. The airplane may be flown, provided the pilot has an ATP rating
C. The airplane may be flown, provided the autopilot and yaw damper systems are operating
D. The airplane must not be flown

22. The switch used to provide a power source for the stick puller system is the:

A. A stick puller is not installed in this airplane
B. STICK PULLER switch
C. L STALL WARN switch
D. R STALL WARN switch

CHAPTER 16
AVIONICS

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ILLUSTRATIONS

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CHAPTER 16 AVIONICS

INTRODUCTION

This chapter describes the avionics installed in the Learjet 60. The avionics system is a completely integrated flight instrument, autopilot, and navigation system. The flight instruments (EFIS) use 7 X 6 inch multicolor electronic displays that combine the functions of an ADI, HSI, MFD, altimeter, airspeed indicator, and vertical speed indicator in a pair of integrated displays. An IAPS (integrated avionics processor system) is included to provide the avionics management and flight control functions. A dual sensor package is installed that includes attitude heading reference systems (AHS), air data computers (ADC), and NAV/COM/pulse radios. The standard radar installation is the Collins WXR-840 radar, with the TWR-850 as an option.

GENERAL

The standard avionics package installed in the Learjet 60 is the Collins Proline 4 System, with options on some of the components. The most significant avionics option, as far as the operator is concerned, is the flight management system (FMS). During initial production, the Collins FMS-850 was installed on all Learjet 60 models; and then, after fourteen aircraft had been produced, the Universal UNS-FMS became available, as a substitute for the Collins FMS-850. Additional universal FMS models are now available/compatible for Learjet 60 aircraft. This manual will discuss only the Universal models of the FMS. The Collins FMS information will be available in a separate supplement.

The UNS-FMS (all models) is referred to as an external navigator when integrated with the Collins Pro Line 4 and does not provide all the functions that the FMS-850 CDU does in the Pro Line 4 system; therefore, when dual UNS-FMSs are installed, two EFIS radar panels (ERP) are added to provide control of some functions that are not controllable with the UNS-FMS CDUs. The functions of the Universal UNS-FMS CDU/ERP, that control the EFIS displays (PFDs and MFD), radio tuning, and the radar control, are described in this chapter (Chapter 16). For more information on the navigation functions of the FMS, refer to the *UNS-FMS Operational Manual*.

A number of other options exist including dual FMSs, dual ADFs, dual Radio Altimeters, turbulence weather radar, EGPWS, MLS, TCAS, LORAN, and GPS. Not all of the optional equipment is described in this chapter.

The avionics package in the Learjet 60 is a versatile but complex system. For pilots more familiar with conventional instruments, it will require a lot of study and practice to become comfortable using all of the capabilities of this system. However, with a little study of the displays and controls, it will be fairly easy to operate the system in the basic modes, which are equivalent to what most conventional instruments provide. As you start to gain experience with the basic modes of operation, you can start using more advanced techniques and more fully utilize the system's capabilities.

This chapter starts with a list of acronyms and abbreviations (Table 16-1), most of which are for equipment that is unique to the modern glass cockpit in the Learjet 60. This list, along with the cockpit layout (Figure 16-1), is a good place to start with familiarization on avionics nomenclature and the various control panel locations in the cockpit.

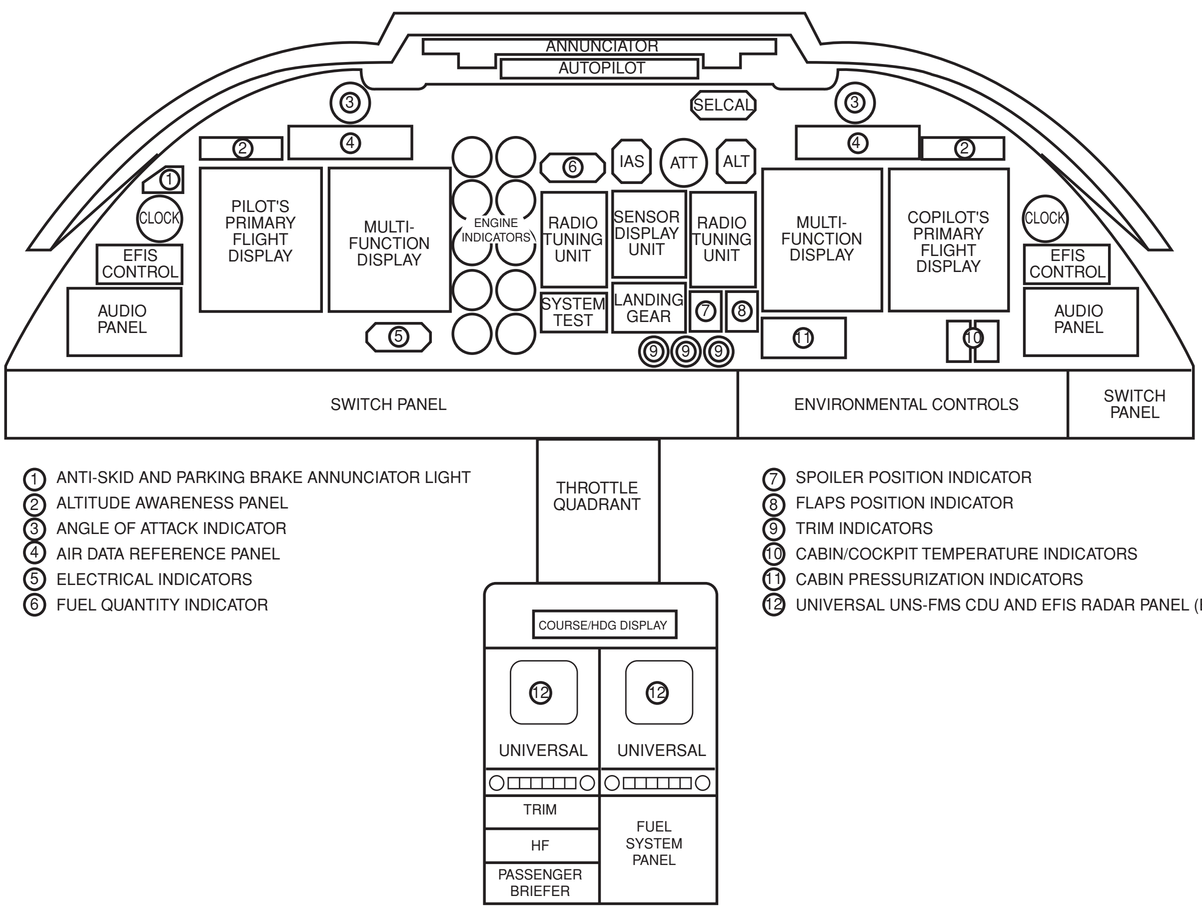


Figure 16-1. Typical Cockpit Layout

Table 16-1. ACRONYMS AND ABBREVIATIONS

KEY	MEANING	KEY	MEANING
AAP	Altitude Awareness Panel	FMS	Flight Management System
AC	Alternating Current	FPD	Flat Panel Display
ACP	Audio Control Panel	GA	Go-Around
ADC	Air Data Computer	GCU	Generator Control Unit
ADF	Automatic Direction Finder	GPS	Global Positioning System
ADG	Air-Driven Generator	GPWS	Ground Proximity Warning System
ADS	Air Data System	GPU	Ground Power Unit
AFCS	Automatic Flight Control System	HCU	Hydraulic Control Unit
AFM	*Airplane Flight Manval*	HF	High Frequency
AGL	Above Ground Level	HFCU	Hydromechanical Fuel Control
AHC	Attitude Heading Computer	HSI	Horizontal Situation Indicator
AHS	Attitude Heading Reference System	IAPS	Integrated Avionics Processor System
AM	Amplitude Modulation	ICPI	Integrated Cabin Pressure Indicator
AGB	Accessory Gearbox	IDS	Instrument Display System
AP	Autopilot	IOC	Input/Output Concentrator
APM	Aircrew Program Manager	ISA	International Standard Atmosphere
APR	Automatic Performance Reserve	ITT	Interstage Turbine Temperature
APU	Auxiliary Power Unit	LH	Left Hand
ARP	Air Data Reference Panel	LOFT	Line Oriented Flight Training
ATP	Airline Transport Pilot	LWD	Left Wing Down
BCTC	Bombardier Customer Training Center	MB	Marker Beacon
BITE	Built-in Test Equipment	MCR	Maximum Cruise
BOV	Bleed-off Valve	MCT	Maximum Continuous Thrust
CB	Circuit Breaker	MEL	Minimum Equipment List
CCU	Cockpit Control Unit	MFD	Multifunction Display
CDC	Cabin Display Computer	MSP	Mode Select Panel
CDU	Control Display Unit	MSW	Control Wheel Master Switch
CHP	Course and Heading Panel	ND	Navigation Display
CPC	Cabin Pressure Controller	OAT	Outside Air Temperature
CPCS	Cabin Pressure Control System	PBE	Protective Breathing Equipment
CRT	Cathode-Ray Tube	PF	Pilot Flying
CVR	Cockpit Voice Recorder	PFD	Primary Flight Display
DAU	Data Acquisition Unit	PMA	Permanent Magnet Alternator
DBU	Data Base Unit	PM	Pilot Monitoring
DC	Direct Current	*PTS*	*Practical Test Standards* (FAA)
DH	Decision Height	RA	Resolution Advisory
DME	Distance Measuring Equipment	RAT	Ram Air Temperature
DTU	Data Trasfer Unit (UNS Disk Drive)	RB	Rudder Boost
EADI	Electronic Attitude Direction Indicator	RH	Right Hand
ECP	EFIS Control Panel	RSS	Radio Sensor System
EEC	Electronic Engine Control	RTU	Radio Tuning Unit
EDP	Engine-Driven Pump	RVDT	Rotary Voltage Differential Transmitter
EDS	Engine Diagnostic System	RWD	Right Wing Down
EDU	Engine Diagnostic Unit	SAT	Static Air Temperature
EFD	Electronic Flight Display	SDD	Sensor Display Driver
EFIS	Electronic Flight Instrument System	SDU	Sensor Display Unit
EGT	Exhaust Gas Temperature	SIC	Second in Command
EGPWS	Enhanced Ground Proximity Warning System	SOV	Shutoff Valve
EICAS	Engine Indicating and Crew Alerting System	SPPR	Single-Point Pressure Refueling
ELT	Emergency Locator Transmitter	TA	Traffic Advisory
EPM	Electrical Power Monitor	TCAS	Traffic Alert and Collision Avoidance System
ERP	EFIS Radar Panel	TCF	Terrain Clearance Floor
ESIS	Electronic Standby Instrument System	TFC	Traffic
ESU	Electronic Sequence Unit	TLA	Thrust Lever Angle
EXT	External	TO	Takeoff
FADEC	Full Authority Digital Engine Control	TR	Thrust Reverser
FCC	Flight Control Computer	USB	Upper Side Band
FCP	Flight Control Panel	VA	Volt Ampere
FCS	Flight Control System	VAC	Volts Alternating Current
FCU	Fuel Control Unit	VDC	Volts Direct Current
FD	Flight Director	VHF	Very High Frequency
FDU	Flux Detector Unit	VNAV	Vertical Navigation
FMA	Flight Control System Mode Annunciator	WXR	Weather Radar
FMC	Flight Management Computer	YD	Yaw Damper

AUDIO CONTROL SYSTEM

The audio control system is used to select the desired audio inputs for broadcast through the speakers or headphones. The audio control system is also usd to select the desired transmitter to which microphone inputs will be directed. A separate audio control system is provided for pilot and copilot. Each system consists of an audio amplifier and audio control panel. The audio control system operates on 28 VDC supplied through the 5-amp L and R AUDIO circuit breakers on the pilot and copilot CB panels respectively. The audio control systems will operate during EMER BUS mode.

AUDIO CONTROL PANEL

An audio control panel (Figure 16-2) is installed at the outboard end of the pilot and copilot instrument panel. Each panel provides the controls necessary to direct audio signals and adjust volume levels. Each panel is used in conjunction with the on-side microphone, headphone and cockpit speaker.

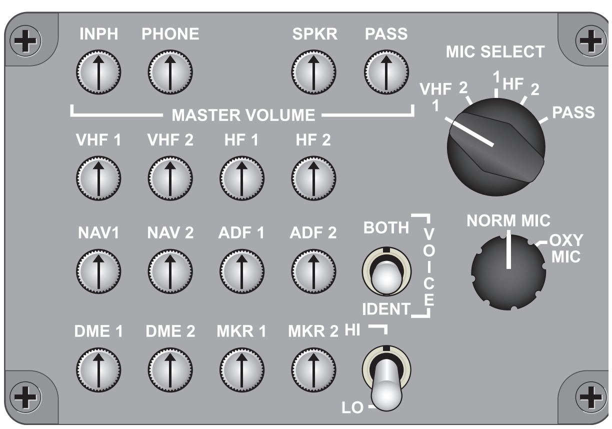


Figure 16-2. Pilot/Copilot Audio Control Panel

AVIONICS MASTER SWITCH

Two avionics master switches are installed which allow the crew to turn groups of avionics equipment off and on through the use of two switches. The LEFT MASTER switch is installed on the pilot avionics panel and the RIGHT MASTER switch is installed on the copilot avionics panel. Table 16-2 shows a list of equipment powered through the master switches. Additional optional equipment may also be powered through the master switches.

Table 16-2. AVIONICS MASTER SWITCH

LEFT MASTER	A/C equipped with Collins FMS	A/C equipped with UNS-1B FMS	RIGHT MASTER	A/C equipped with Collins FMS	A/C equipped with UNS-1B FMS
ADF 1	•	•	COMM 2	•	•
NAV 1	•	•	NAV 2	•	•
DME 1	•	•	DME 2	•	•
ATC Transponder 1	•	•	ATC Transponder 2	•	•
PFD 1	•	•	PFD 2	•	•
MFD 1	•	•	MFD 2	•	•
AAP 1	•	•	AAP 2	•	•
FD 1 (AP/YD/RB)	•	•	FD 2 (AP/YD/RB)	•	•
IAPS TEMP Control 1	•	•	IAPS TEMP Control 2	•	•
HF 1	•	•	RTU 2	•	•
SELCAL	•	•	Cabin Display	•	•
Radio Altimeter	•	•	Radar	•	•
ERP 1	N/A	•	ERP 2	N/A	•
FMS 1	•	①	FMS 2 (if installed)	•	①
CDU 1	•	①	CDU 2	•	①
OSS 1 & GPS 1	•②	①	OSS 2 & GPS 2 (if installed)	•②	①

① *60-001 thru 60-057 not incorporating SB 60-34-2.*
② *60-004 thru 60-009 not incorporating ECR 3589 or equivalent, the long range nav sensors are not on the avionics master.*

VHF COMM 1, the left RTU, and the SDU are not controlled by the avionics master switches. They are powered by emergency bus power.

Both avionics master switches must be on for the autopilot, yaw damper and rudder boost (nosewheel steering force switches) to operate.

DC electrical power is provided to the avionics master switches through the L and R AVIONICS MASTER circuit breakers, on the left and right circuit-breaker panels, respectively.

EFIS CONTROL PANEL (ECP)

An EFIS control panel (ECP) (Figure 16-3) is installed on both the pilot and copilot instrument panels, outboard of the primary flight displays (PFD). The switches on the ECPs are mechanical alternate action switches. On-side selection is indicated when the switch label is green and cross-side selection is indicated when the switch label is amber.

Figure 16-3. EFIS Control Panel (ECP)

The pilot ECP has priority on the first three switches (AHS, ADC, and CDU). Both pilots cannot select cross-side at the same time. If they attempt to, the pilot selection will take priority and the copilot will not be able to select the cross-side. If either pilot selects cross-side on one of these first three switches, the cross-side system will be annunciated on both PFDs. For example, if the pilot selects the cross-side AHS, ATT 2 and MAG 2 will be annunciated on both PFDs. Also, when AHS cross-side is selected on either side, MAG 1 or MAG 2 (depending on which side selects AHS cross-side) will appear on the MFDs, if heading information is being displayed on the MFDs at the time. When both sides have on-side systems selected, this annunciation does not appear on the PFDs or MFDs. The cross-side mode annunciation is described further in the AHS, ADC, and CDU parts of this chapter.

The function of each ECP switch will not be described at this point, but instead, will be described along with the systems they affect. For example, the AHS switch is covered under attitude heading reference system (AHS) in this chapter, the ADC switch is covered under air data system (ADS), etc.

Aircraft equipped with UNS flight management systems will not have a OFF–RMT TUNE switch on the ECP. Instead, the switch will be labeled SPARE.

Electrical DC power for the ECPs is provided through the EFIS CTL 1 and EFIS CTL 2 circuit breakers, on the left and right CB panels, respectively. The EFIS control panels are not powered in the emergency bus mode of operation.

ALTITUDE AWARENESS PANEL (AAP) AND AIR DATA REFERENCE PANEL (ARP)

An AAP and an ARP are located above each PFD. The function of these two panels is to set altitude and air data references on the PFD displays. The AAP is used to set minimum descent altitude (MDA), reporting altitude (RPT) and decision altitude (DH) on the PFD altitude display. The ARP is used to set takeoff speeds (V_1, V_R, and V_2), the IAS reference bug, vertical speed, altimeter setting and altitude preselect. Operation of the AAP and ARP panels is described under primary flight display (PFD) in this chapter.

On aircraft equipped with the UNS FMS, electrical DC power for the AAPs is provided through the ERP-AAP 1 and ERP-AAP 2 circuit breakers. Electrical DC power for the ARPs is provided through the ADC-ARP 1 and ADC-ARP 2 circuit breakers, on the left and right circuit-breaker panels, respectively. The AAPs and ARPs are not powered and do not function in the emergency bus mode of operation.

AIR DATA SYSTEM (ADS)

GENERAL

Dual ADC-850D air data computers are installed to provide, redundant, air data information. The two digital air data computers are functionally and physically isolated from each other. Each air data computer has three, redundant, low-speed output buses. The first bus goes to the on-side large displays for primary air data information. The second bus is connected to the on-side AHS computer to provide TAS data, and to the cross-side large display for cross-side air data display. The third bus is connected to the IAPS for use by other systems that use air data parameters.

AIR DATA COMPUTER OPERATION

The ADCs are powered whenever DC power is applied to the aircraft's electrical system. There are no power ON–OFF switches in the cockpit. The dual air data computers provide parallel but independent sources of data for pilot and copilot instruments and other systems.

Normally ADC 1 will be selected to provide air data to the pilot instruments and ADC 2 will be selected to provide air data to the copilot instruments (on-side). The ADC switches on the left and right EFIS control panels will be illuminated in green when the ADCs are selected to on-side.

There is no computer fail light to indicate computer failure, but if an ADC fails, it will become apparent, by flags over the airspeed, altitude and vertical speed scales, and loss of temperature (SAT) indication on the on-side PFD. If an ADC fails, the remaining ADC can provide air data information to both sets of flight instruments. To restore air data to the failed side, the pilot with the failed ADC should depress the ADC switch on the on-side EFIS control panel (see Figure 16-3). If the pilot selects cross-side ADC on the left ECP, the ADC switch label will turn to amber and an ADC 2 annunciator will appear on both PFDs. ADC 1 will appear on both PFDs if ADC cross-side was selected on the right side of the cockpit. Cross-side ADC can only be selected on one ECP at a time, as explained earlier under EFIS control panel (ECP).

If the autopilot is coupled to the side using cross-side ADC, all air data modes will synchronize at engagement and cannot be changed using the on-side ARP. Autopilot function can be restored if the autopilot control is moved to the side that has the good ADC. The transponder must also be selected on the side of the good ADC, to regain altitude reporting (Mode C).

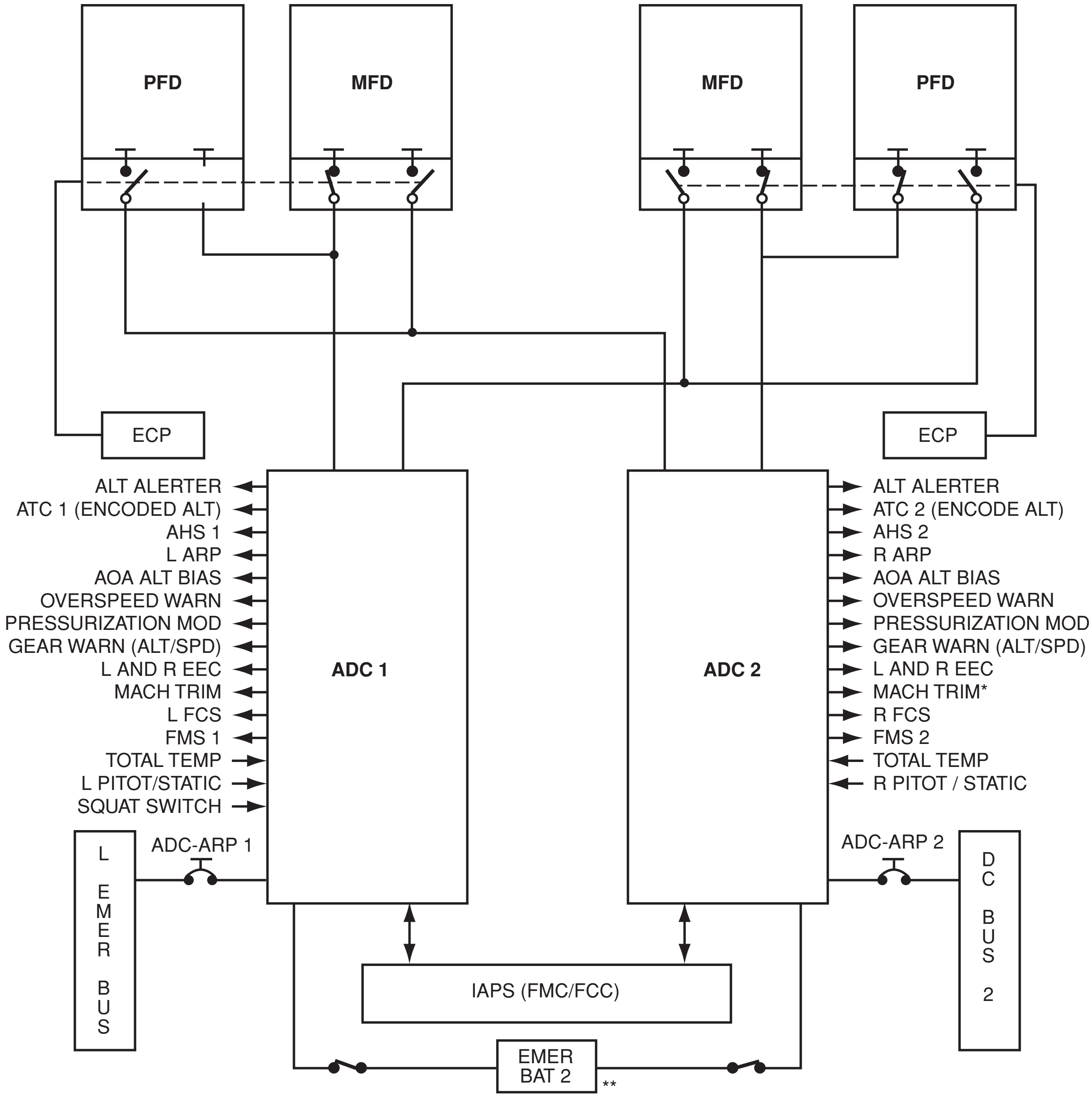
Some systems (gear warning system, cabin pressurization controller, V_{MO}/M_{MO} warning horn, etc.) receive data from both ADCs for redundancy, and continue to function with single ADC failure (Figure 16-4). The Mach trim computer normally receives airspeed information from ADC 1, but if the pilot has selected ADC cross-side, airspeed information for Mach trim operation will be provided by ADC 2.

In the event of a dual ADC failure or loss of electrical power, the pilot must use the standby Mach/airspeed indicator and standby altimeter located in the center of the instrument panel. Refer to Section 4, Abnormal Procedures, in the *AFM* for single or dual ADC failure.

ADC ELECTRICAL POWER

The No. 1 ADC is supplied with DC electrical power through the ADC-ARP 1 circuit breaker, located on the pilot CB panel (DC emergency bus). The No. 2 ADC is supplied DC electrical power through the ADC-ARP 2 circuit breaker, located on the co-pilot CB panel (DC Bus 2). ADC 1 and 2 also have a secondary power source from EMER BAT 2 when the EMER BUS switch is NORMAL. This secondary power source is primarily in place to ensure the ADCs remain powered during engine start or during other conditions which may require a temporary power source to the ADCs.

With a dual generator failure, after the EMER BUS switch is placed to EMER, the ADC No. 1 and No. 2 are powered only by the EMER BUS. This is a function of the position of the



* WHEN PILOT HAS ADC CROSS-SIDE SELECTED

** EMER BAT 2 PROVIDES BACKUP POWER TO THE ADCs ONLY WHEN EMER BUS SWITCH IS IN THE NORMAL POSITION

Figure 16-4. ADC System Schematic

EMER BUS switch. If EMER BUS voltage is depleted, ADC No. 1 and No. 2 will no longer receive power. Therefore, the EMER BAT voltage will not be utilized to power the ADCs during this critical phase of flight.

PITOT-STATIC SYSTEM

GENERAL

A dual pitot-static system serves the ADCs. Two pitot probes, one on each side of the aircraft nose, supply impact pressure, and two static ports on each pitot probe provide static pressure to the ADCs. A third pitot probe, mounted above the main probe on the right side of the aircraft, provides impact pitot pressure and static pressure to a standby Mach/airspeed indicator and standby altimeter. Moisture drains are provided in the pitot and static lines.

PITOT PROBES/STATIC PORTS

Each pitot-static (Figure 16-5) probe has a pitot port in the tip and two static ports (Static 1 and Static 2). The probes contain electrical heating elements for anti-icing. Current to the heating elements is controlled by the L and R PITOT HEAT switches. The left pitot heat switch controls heating to the left pitot-static probe, and the right pitot heat switch controls pitot heat for both probes, on the right side of the aircraft (standby pitot-static probe and right main system pitot-static probe). See Chapter 10—"Ice and Rain Protection," for more information on pitot heat.

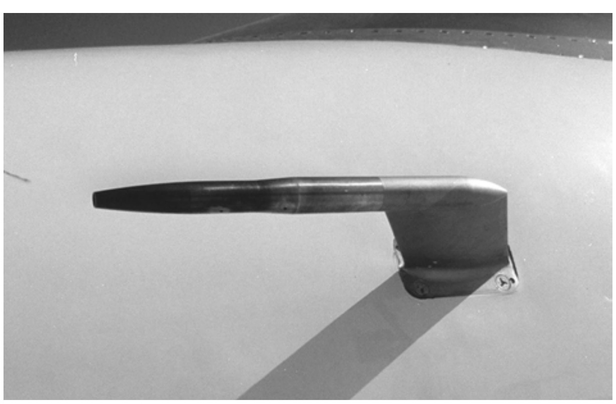


Figure 16-5. Pitot-Static Probe

The pitot systems (Figure 16-6) are independent. The left pitot probe provides pitot source air for ADC 1, and the right pitot probe provides pitot source air for ADC 2. There are four static sources in the main pitot-static system, two on each pitot-static probe. The static sources (Static 1 and Static 2) each sense static pressure, through two holes in the probe (one on top of the probe and one on the under side). Under normal conditions (static source switch in BOTH position) Static 1 on each pitot-static probe is connected to Static 2 on the opposite pitot-static probe. This is to reduce system error.

The source of static pressure is controlled with the STATIC SOURCE switch, located on the pilot lower outboard switch panel. The STATIC SOURCE switch is a horizontal toggle-type with three positions: L (left), BOTH (center), and R (right) (Figure 16-7). This switch is normally set to the BOTH position except in the event one of the pitot-static probes becomes inoperable or unreliable. When the switch is set to L or R, solenoid operated shutoff valves are energized to shut off the static sources from the opposite side static ports (Figure 16-6).

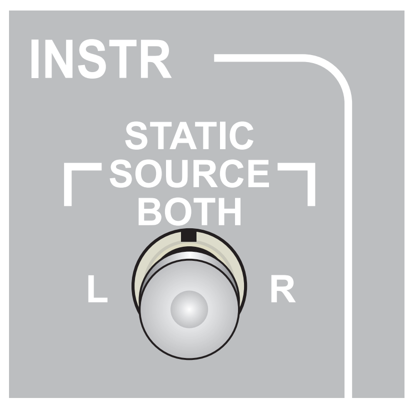


Figure 16-7. STATIC SOURCE Switch

When the STATIC SOURCE switch is in the L position, static pressure is provided to both ADCs from Static 1 and Static 2 of the left pitot-static probe; and in the R position, static pressure is provided to both ADCs from Static 1 and Static 2, of the right pitot-static probe.

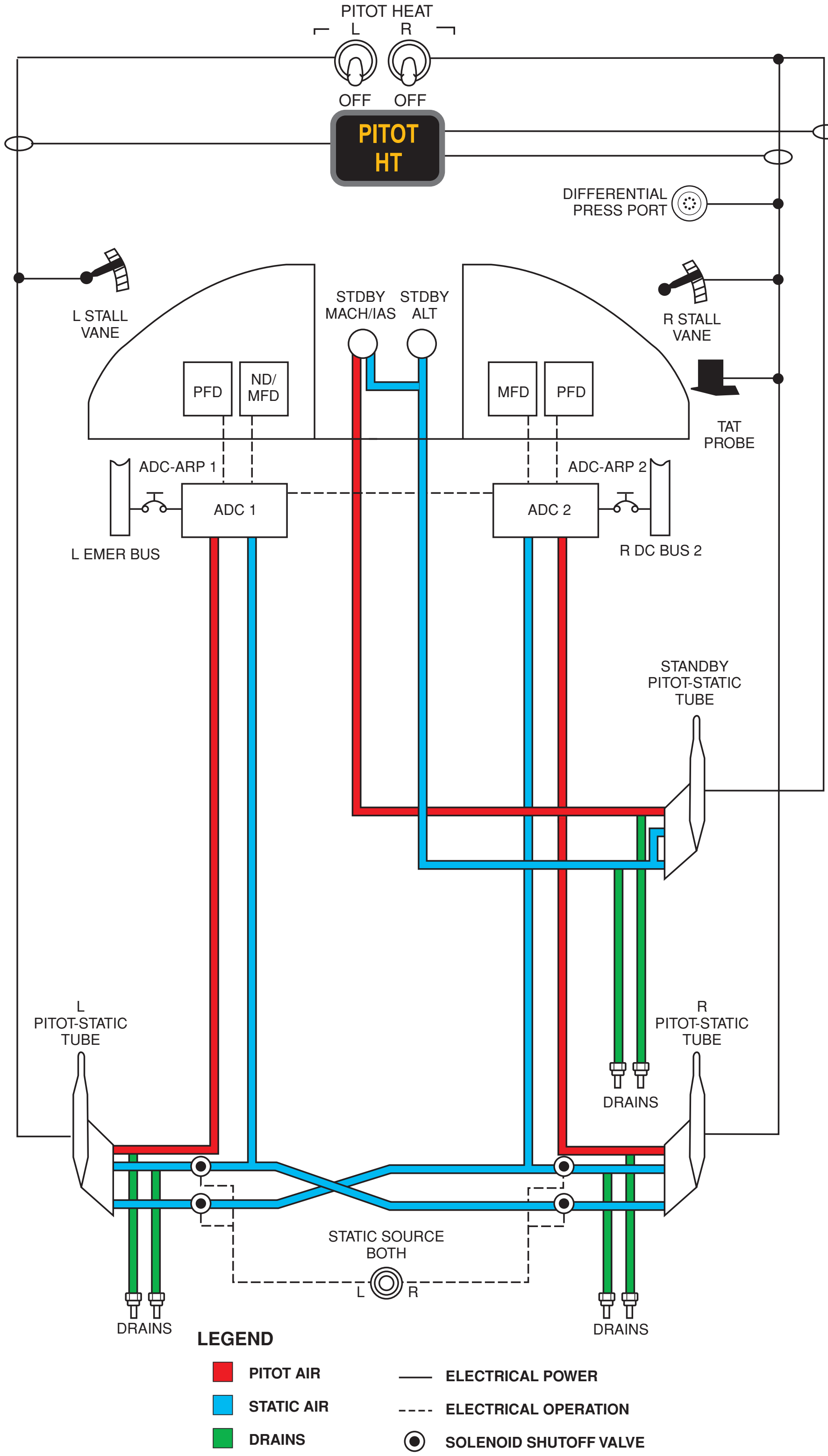


Figure 16-6. Pitot-Static and Air Data System Schematic

The solenoid shutoff valves operate on 28 VDC supplied through the STATIC SOURCE circuit breaker, on the right circuit-breaker panel.

Refer to Section 4, Abnormal Procedures, in the *AFM* for Pitot Static System Malfunction.

MOISTURE DRAINS

Six moisture drains are provided in the pitot and static lines. Two push up to drain valves are located outboard of the left side nose wheel door to drain the left main pitot-static system (Figure 16-8). Four push up to drain valves are located outboard of the right side nose wheel door to drain the standby pitot-static system (two forward) and the right main pitot-static system (two aft).

Figure 16-8. Pitot-Static Drain Valves

STANDBY PITOT-STATIC SYSTEM AND INSTRUMENTS

A standby Mach/airspeed indicator and barometric altimeter (Figure 16-9) are mounted on the center instrument panel, on either side of the standby attitude indicator, to provide backup in the event of aircraft electrical failure or dual air data system failure.

These instruments are lighted with electrical power from the No. 1 emergency battery in the event of aircraft electrical system failure. There are several different configurations available for the standby instruments.

Standby Attitude Indicator (Aircraft 60-001 thru 60-248)

Two-Inch Indicator

A standard 2-inch standby attitude indicator (Figure 16-10) may be installed on the center instrument panel. The indicator will provide 92º of climb, 78º of dive, and 360º of roll attitude information. A sky pointer is incorporated to indicate vertical in any roll attitude. Roll index marks at 10, 20, 30, 60 and 90º provide measurement of angular displacement from vertical as indicated with the sky pointer. The two-colored pitch drum is directly linked to the spin motor of the gyro to provide direct reading of aircraft attitude in both pitch and

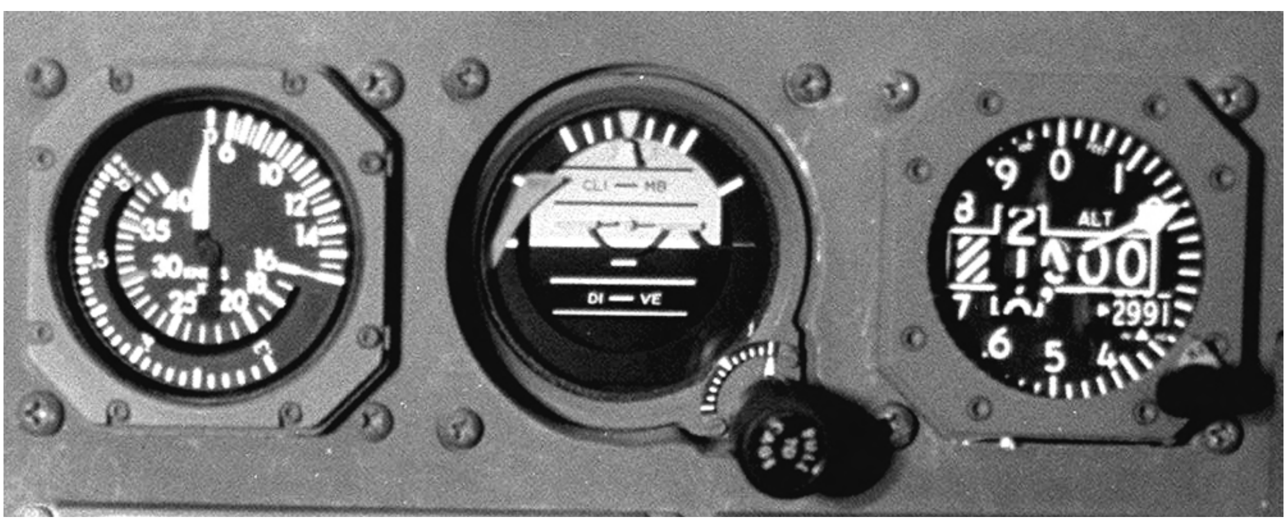


Figure 16-9. Standby Mach/IAS Indicator, Attitude Indicator, and Altimeter

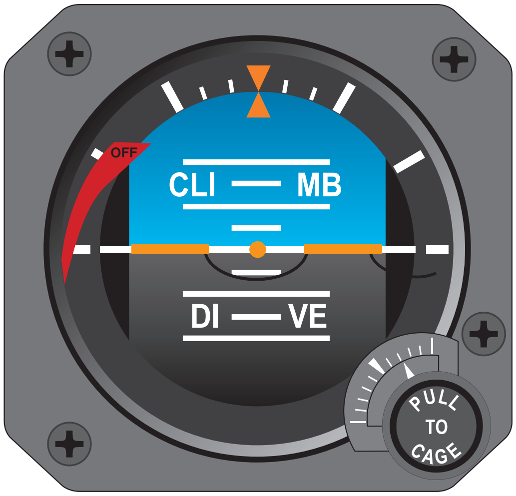


Figure 16-10. Standby Attitude Indicator

roll. The light colored area marked CLIMB represents the sky. The dark colored area marked DIVE represents the earth. A horizontal white line divides the colored areas to represent the horizon. Index marks are incorporated on the drum to indicate every 5º of pitch in both CLIMB and DIVE attitudes.

The adjustable miniature airplane symbol indicates aircraft pitch and roll attitude relative to the horizon. The symbol is adjustable through 5º of pitch in both CLIMB and DIVE directions using the PULL TO CAGE knob. Rotating the knob moves the symbol and a pointer that indicates symbol pitch displacement on a scale marked in 1º increments. The knob is slowly pulled out and rotated clockwise to cage the gyro. When released, the knob will remain in the extended position. A red OFF flag will appear if the gyro is caged, power is not applied or was lost or the gyro became inoperative.

The standby attitude indicator is powered by 28 VDC supplied by EMER BAT 1. This indicator is used to cross check the pilot and copilot attitude displays and serves as a backup unit in the event both AHS 1 and AHS 2 become inoperative.

Three-Inch Indicator

An optional J.E.T. ADI-331 standby attitude indicator may be installed in place of the standard 2-inch indicator (Figure 16-11). It is a 3-inch, self-contained gyro powered by 28 VDC supplied from EMER BAT 1. This unit incorporates all of the features described for the 2-inch unit as well as crosspointers to show deviation from glideslope and localizer beams. If the signal becomes invalid, GS and/or LOC flags will come into view and the associated crosspointer(s) will be stowed. The crosspointers and flags are active only when an ILS frequency is tuned on the #1 NAV receiver. An inclinometer, on the front of the instrument, shows aircraft slip/skid information.

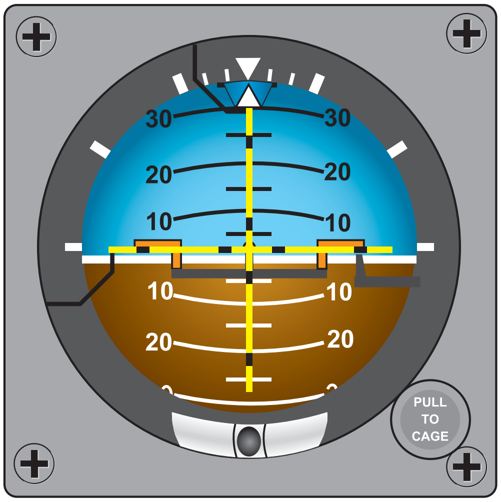


Figure 16-11. Three-Inch Indicator

The ADI-331 installation is designed to provide attitude, glideslope and localizer information independent of the normal electrical system. In the event the normal electrical system fails, EMER BAT 1 will supply 28 VDC to power the standby attitude indicator, and EMER BAT 2 OR EMER BAT 1 will supply 28 VDC to power the #1 NAV receiver and the #1 radio tuning unit (RTU). Although the #1 RTU will be operative under these conditions, its only meaningful function will be to tune the #1 NAV receiver to an ILS frequency since the COM, ADF, and ATC equipment will be inoperative.

Electronic Standby Instrument System (Aircraft 60-001 thru 60-248)

An optional electronic standby instrument system (ESIS) (Figure 16-12) is available and replaces the standby attitude, altimeter and airspeed/Mach indicators located on the center instrument panel. This single LCD indicator provides the pilot and copilot with pitch and roll attitude, slip/skid information, baro altitude, airspeed, Mach, dual baro-set, V_{MO}/M_{MO}, localizer and glideslope deviation and marker beacon annunciation.

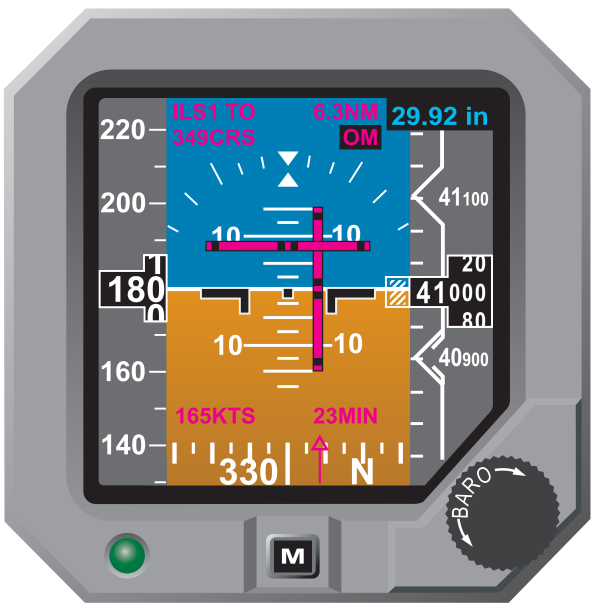


Figure 16-12. Electronic Standby Instrument System Indicator

The system consists of a remote air data computer and an ESIS display indicator. For a detailed description of this system, refer to the current *BF Goodrich Aerospace GH-3000 Electronic Standby Instrument System Pilot's Guide*.

Remote Air Data Computer

The remote air data computer receives pitot and static pressure from the standby pitot-static system and provides airspeed and altitude information to the display indicator. It is powered by EMER BAT 1.

ESIS Display Indicator

The ESIS display indicator presents information on a color active matrix liquid crystal display. The display format resembles that of the primary flight displays (PFDs). The indicator is powered by EMER BAT 1.

Integrated Standby Indicator System (ISIS) (Aircraft 60-249 and subsequent)

An integrated standby indicator system (ISIS) (Figure 16-13) is located on the center instrument panel. This indicator is a Smiths Industries solid state, graphic display standby indicator system. It consists of a self-sensing single box unit and is powered by 28 VDC supplied by EMER BAT 1. Power is also provided to RTU 1 and NAV 1. This single LCD indicator provides the pilot and copilot with pitch and bank angle, slip/skid indications, altitude, airspeed, Mach number, dual baro-set, and V_{MO}/M_{MO} indications. Localizer and glideslope deviation is provided if NAV 1 is tuned to an ILS. It is designed to mimic the primary EFIS system.

Figure 16-13. Intergrated Standby Indicator System

For a more detailed description of this system, refer to the current Smiths Industries *Integrated Standby Indicator System Pilot's Guide* (P/N CDK316).

ATTITUDE HEADING REFERENCE SYSTEM (AHS)

GENERAL

The Learjet 60 has dual AHS-85 attitude heading reference systems. Each AHS consists of an AHC-85E attitude heading computer, internal compensation unit (ICU) and flux detector unit (FDU). Each system independently senses the aircraft attitude, heading and three-axis rate/accelerations.The AHSs replace conventional vertical gyros, directional gyros, rate gyros and linear accelerometers.

ATTITUDE HEADING COMPUTERS (AHC)

The AHCs are located in the right nose, avionics bay, with the internal compensation units (ICU), fastened to the front of the AHCs. The ICU allows the computer to be adjusted for the specific aircraft and it should remain with the aircraft if an AHC is changed out. A FDU for each computer is located in a dry bay area—one in each wingtip.

There are two multisensor wheels (Figure 16-14) in each AHC-85 that rotate about axes fixed 90° to each other. These multisensors remain fixed in relation to the aircraft axes. They do not remain rigged in space as do conventional gyros. The multisensors have piezoelectric crystals attached at different

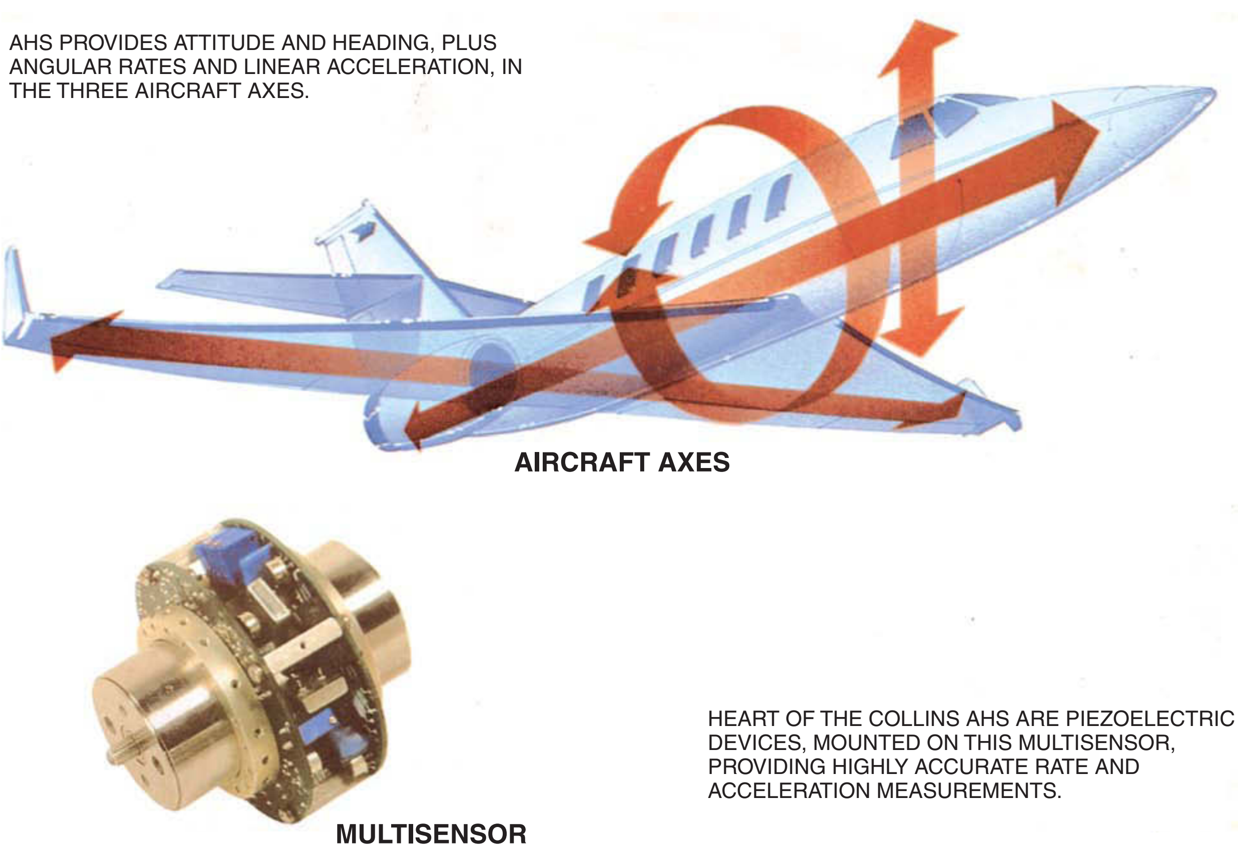


Figure 16-14. Attitude Heading Reference System (AHS) Multisensor

angles. The crystals flex with accelerations, which changes the electrical resistance of the crystal, thereby sensing the acceleration in each axes. The AHCs establish a reference to level during initialization. In this regard, the AHC is similar to an inertial system. The flux sensors allow the heading reference to be slaved to magnetic when the cockpit SLAVE-FREE switch is in the SLAVE position. The ADCs provide TAS and VS inputs to the AHCs (Figure 16-15).

INITIALIZATION

Initialization of the AHSs automatically begins and continues for approximately 70 seconds after power has been applied to the aircraft electrical system (main batteries ON). The ATT flags in the PFDs will be removed when initialization has been completed. During initialization, the aircraft should not be moved or initialization will be delayed. Also, the HEADING SLAVE-

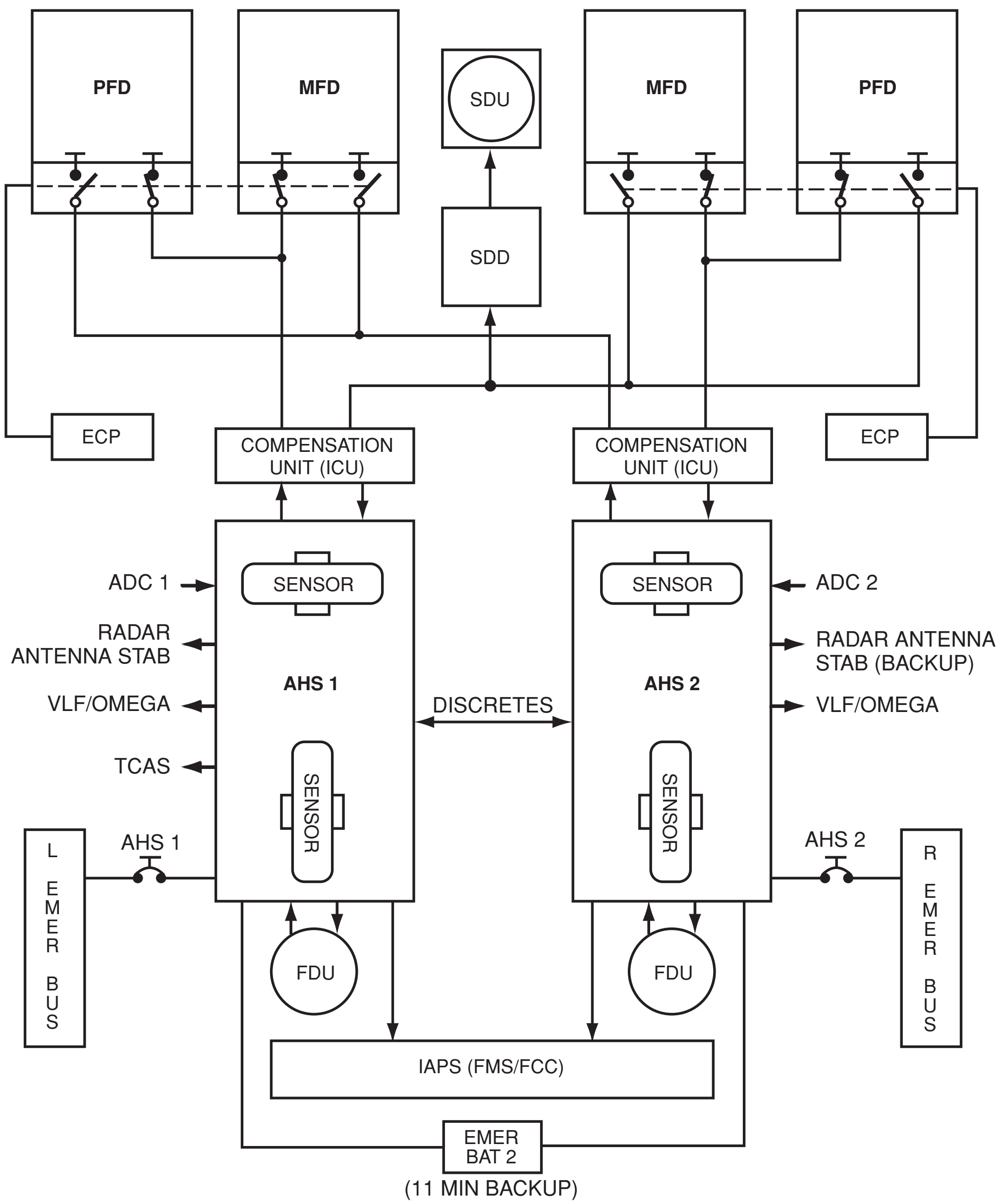


Figure 16-15. AHS System Schematic

FREE switches should be in the SLAVE position. With the SLAVE-FREE switch in the FREE position, initialization can take up to 10 minutes.

OPERATION

The computers provide all attitude and heading measurements, as well as angular rates and linear acceleration. This information is utilized by the electronic flight instruments, autopilot, and yaw damper.

Normally AHS 1 is selected to provide attitude and heading information to the pilot instruments and AHS 2 is selected to provide attitude and heading information to the copilot instruments. The AHS switches, on the left and right EFIS control panels (ECP) (see Figure 16-3), will be illuminated in green when the AHSs are selected to on-side.

Only AHS 1 can provide heading information to the sensor display unit (SDU).

If an AHS fails, the cross-side AHS can be selected by depressing the green AHS switch on the failed side ECP. If the green AHS switch is depressed on the left ECP, for example, the green AHS will extinguish and an amber AHS on the switch will illuminate. Also, yellow ATT 2 and MAG 2 annunciators will appear on both PFDs. Yellow MAG 2 annunciators will also appear on the MFDs if heading information is being displayed on them at the time (HSI mode).

If AHS cross-side is selected on the copilot ECP (assuming the pilot has on-side selected), the AHS will turn amber on the copilot ECP and yellow ATT 1 and MAG 1 annunciators will appear on both PFDs. Yellow MAG 1 annunciators will also appear on the MFDs if heading information is being displayed on them. AHS cross-side cannot be selected on both ECPs at the same time; if attempted, the left side will have priority and AHS 2 information will be presented on both sides.

If the on-side AHS loses power or input data to the PFD fails for any reason, red, boxed ATT and MAG flags will be displayed on the PFD and the attitude and heading displays will be removed. If AHS 1 fails, heading information to the SDU will also be lost and HDG in a box with an X through it will be displayed at the top of the compass card.

The autopilot uses inputs from both AHSs, and if either fails, the autopilot and yaw damper will both be inoperative.

If the left or right side AHS fails, see attitude heading system (AHS) Malfunction in the Abnormal Procedures section of the *AFM*.

COMPARATOR

The pitch, roll and heading information from the AHSs is monitored by a comparator system in the EFIS.

In the event AHS heading information disagrees by more than 6° plus one half of the roll angle, or pitch or roll disagrees by approximately 4° (3° if glideslope captured), a corresponding yellow, boxed HDG, PIT or ROL annunciator will illuminate on both PFDs. A yellow, boxed HDG will also appear on the MFDs if heading information is being displayed on them. These annunciators will remain in view until the mismatch is corrected or both pilots select the same AHS on their ECPs.

It may be necessary to refer to the standby attitude indicator or the standby magnetic compass to determine which AHS is in error. See Comparator Warning in the Abnormal Procedures section of the *AFM*.

The comparator system also compares radio altimeter, glideslope and localizer information on those aircraft certified for Category II approaches.

AHS ELECTRICAL POWER SOURCE

The AHSs receive DC electrical power as soon as electrical power is applied to the aircraft. They are powered through the AHS 1 and AHS 2 circuit breakers, on the left and right side

circuit-breaker panels, respectively. The AHSs continue to receive power during emergency bus mode of operation. AHS 1 and AHS 2 normally receive electrical power from the left and right emergency buses, respectively. If power from either bus is interrupted or if there is low voltage from the bus, emergency battery 2 will automatically provide power to the affected AHS for 11 minutes.

COMMUNICATIONS

VHF COMMUNICATIONS

Dual VHF communications transceivers and controls are installed to provide AM voice communication capability.

Several VHF COMM packages are available as shown in Table 16-3.

The transceivers are SELCAL compatible with analog audio interfaces. Tuning is accomplished via the radio tuning units (RTUs) on the center instrument panel or via the control display units (CDUs) in the pedestal. The design of the system is such that all radio management functions are channeled through the RTUs, regardless of their origin.

The left RTU normally tunes COMM 1 and the right RTU normally tunes COMM 2. If an RTU fails, the remaining RTU is capable of tuning both COMM 1 and COMM 2, provided that the failed RTU is turned off on the EFIS control panel (see Figure 16-3). Power for the system is 28 VDC supplied through the 7.5 amp COMM 1 and COMM 2 circuit breakers on the pilot and copilot circuit breaker panels. COMM 1 is powered during EMER BUS operations.

HF COMMUNICATIONS

An HF (high frequency) communication system is installed to provide long range communication capability. The system operates on any 0.1 kHz frequency between 2.0 and 29.9999 MHz. The system consists of a control/display unit located on the aft center pedestal (Figure 16-16), a remote power amplifier and antenna coupler, remote receiver/transmitter, and antenna. System power is 28 VDC supplied through current limiters and controlled by a remote control circuit breaker. The remote control circuit breaker is controlled by the 0.5 amp HF 1 circuit breaker on the pilot CB panel. The HF receiver is SELCAL compatible.

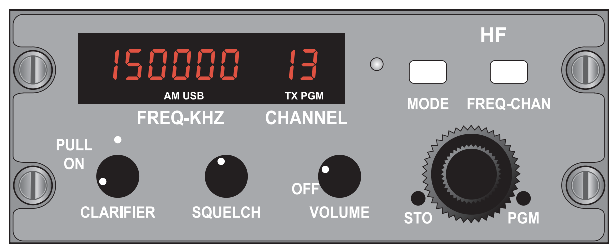


Figure 16-16. Control Display Unit (CDU)

The above is presented in a general manner and is intended for familiarization only. For a detail description and operation of the HF communications system, refer to the appropriate HF operators manual.

Table 16-3. AVAILABLE OPTIONS

AVAILABLE PACKAGE	FREQUENCY RANGE	FREQUENCY SPACING
1	118.00 to 135.975 MHz	25 kHz steps
2	118.00 to 151.975 MHz	25 kHz steps
3	118.00 to 136.975 MHz	8.33 kHz steps
4	118.00 to 136.975 MHz	8.33 kHz steps
	137.00 to 151.975 MHz	25 kHz steps

SELCAL

The SELCAL system permits the selective calling of individual aircraft over normal radio communications circuits linking the ground station with the aircraft. The SELCAL system is integrated into the VHF and HF communication systems to relieve the flight crew from continuously monitoring communications frequencies during flights of extended duration. The systems consists of a decoder unit and the SELCAL control panel (Figure 16-17) on the center instrument panel. The system is powered by 28 VDC through the 1 A SELCAL circuit breaker on the pilot CB panel.

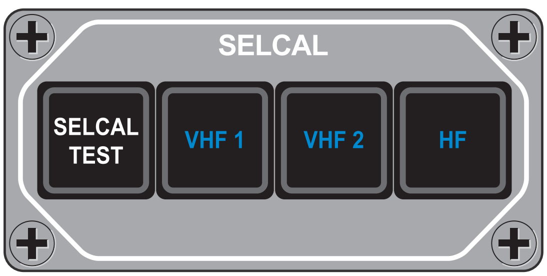


Figure 16-17. SELCAL Control Panel

When a call is received, the appropriate annunciator (VHF 1, VHF 2, HF1 or HF2) in the SELCAL control panel illuminates and an intermittent aural tone sounds. The annunciator is then depressed and communication with the caller can be established. When the button is depressed, the aural tone will cease.

The system may be tested by depressing the SELCAL TEST button. When this is done, all three annunciators illuminate and an aural tone sounds. In order to extinguish the annunciators, each button must be depressed. The aural tone will cease when the last button is depressed.

RADIO SENSOR SYSTEM (RSS)

GENERAL

The radio sensor equipment consists of dual COM, dual NAV, dual DME, dual ATC, single ADF, single RAD ALT, and a single VLF/Omega (with GPS option) receiver. A second VLF/Omega, radio altimeter and ADF receiver may be installed as options. LORAN, MLS and TCAS are also available.

Radio sensor data in the Learjet 60 avionics system follows several independent paths that lead to multiple display devices. One path leads to the integrated avionics processor assembly (IAPS), where the data is placed onto concentrated buses for use by the PFDs, MFDs, and FMS. A second path leads to the SDU (sensor display unit) that is mounted in the center of the instrument panel.

Radio sensors are normally controlled through the radio tuning units (RTUs). The number one radios are normally controlled by the left RTU and the number two radios are normally controlled by the right RTU.

Either CDU (control display unit), located on the center pedestal, can also be used to tune the VHF COM, VOR/LOC, and ADF receivers and set the ATC transponder code. The CDUs do not have the capability to tune the MLS receiver (if installed).

REMOTE RADIO SWITCHES

Each EFIS control panel (ECP) has a remote OFF–RTU switch.

The OFF–RTU switches can be used to disable the on-side RTU. If a RTU should fail or malfunction, the operating RTU can control both sets of radios after the faulty RTU is disabled. Depressing the OFF–RTU switch on the side with the failed RTU will allow both sets of radios to be tuned on the remaining RTU through use of the 1/2 switch on the RTU. Use of the 1/2 switch is described under RTU Function Keys in this section.

RADIO TUNING UNIT (RTU)

GENERAL

The two radio tuning units are located on either side of the SDU in the center of the instrument panel. The RTUs provide the controls for frequency selection, mode of operation, and self-test for the VOR/ILS, ADF, VHF COM, ATC transponder radios and TCAS (if installed). If frequencies are selected by the crew via the CDU, they are sent to the appropriate RTU and then to the proper radio. By doing this, the RTUs will always display the active frequencies. All RTU memory is nonvolatile, so frequencies, modes, etc. are retained during power outages.

The RTU lighting and display brightness is controlled with a brightness knob on the upper-right corner of each RTU and by the EFIS dimmer controls on the L and R INSTR LIGHTS panels.

RTU ELECTRICAL POWER SOURCE

DC electrical power is provided to the RTUs through the RTU 1 and RTU 2 circuit breakers on the respective left and right circuit-breaker panels. The left RTU, AUDIO 1, AUDIO 2, COM 1, NAV 1, ADF 1, and ATC 1 all continue to operate in the emergency bus mode of operation. The right RTU is turned on with the right avionics master switch, whereas the left RTU and COMM 1 are powered as soon as aircraft electrical power is turned on.

RTU CONTROL FUNCTONS

The RTU has two concentric knobs for frequency/code setting, five line-keys, and six dedicated functions (Figure 16-18).

The five line-keys, on the right side of the RTU, are used to select a radio for retuning and to access the mode control pages. Pressing a line-key that does not have the selector star on that line will cause the star to move to that line. The star shows the radio that can be tuned with the frequency/code select knobs (tuning knobs). The selector star is normally displayed on the PRE line, and automatically returns to that line after approximately 15 seconds of inactivity. Pressing a line key a second time (while the selector star is on that line) will display the COM, NAV, ADF, or ATC mode page.

The COM frequency can be changed by depressing the COM line key (star moves to COM line) and then selecting the desired frequency with the tuning knobs. Another way to change the COM frequency is to enter the frequency in the PRE line first and then move it up to the active COM line. This is the more popular choice. To change the COM frequency through the preset, first press the PRE line key to display the selector star on the PRE line. This step isn't necessary if the star is already there. Next, change the frequency using the tuning knobs; then, press the PRE line again to interchange the preset and active COM frequencies. The old frequency is available for instant recall by pressing the PRE line key again. An upward pointing arrow displays to the right of the PRE legend while this line is selected. This arrow indicates that the preset frequency can be moved upward and become the active frequency.

A TX replaces the COM while the radio is keyed for transmissions.

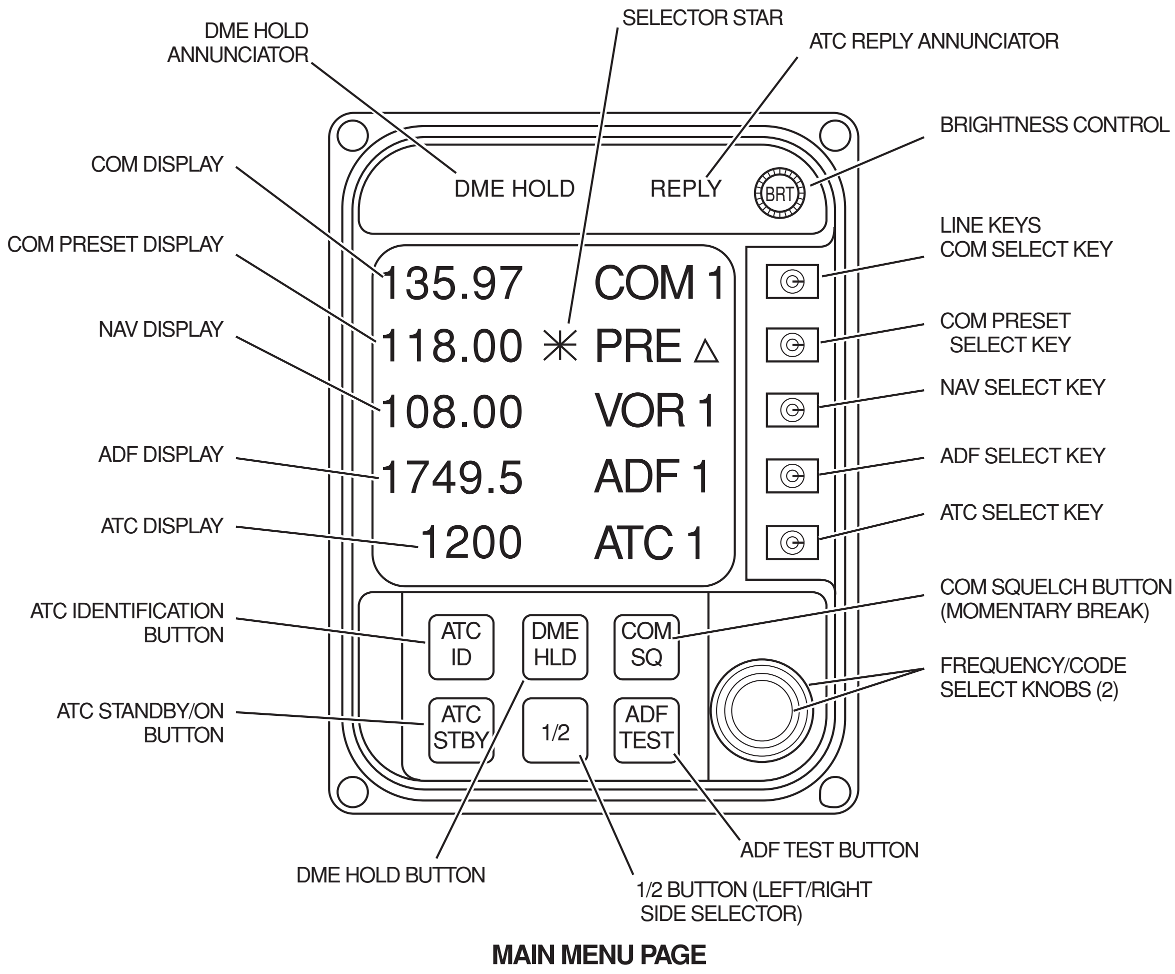
To change the VOR/ILS or ADF frequency or ATC code, you simply depress the desired line key to move the selector star to that line and then retune the frequency or code with the selected knobs.

The ATC line displays only on the RTU that corresponds to the active transponder. Only one transponder can be operating at a time. The active transponder is selected on the left or right EFIS control panel (ECP) (see Figure 16-3). The 1 or 2 annunciator, on the ATC 1/2 switch on each ECP, illuminates to indicate which transponder is selected. If the ATC switch is depressed on either ECP, the transponder number (1 or 2) changes on both ECPs. If ATC 1

is selected on either ECP, the bottom line on the left RTU will indicate STBY or ATC 1 depending on whether the transponder is in standby or ON, and the bottom line on the right RTU will be blank. If ATC 2 is selected on either ECP, the transponder status (STBY or ATC 2) will be displayed on the bottom line of the right RTU.

RTU TUNING KNOBS

The frequency/code select knobs (tuning knobs) consist of two concentric knobs (Figure 16-18). The larger knob changes the left digits and the smaller knob changes the right digits.



NOTE:
THE UPPER-HALF OF THE LEFT-MOST DISPLAY DIGIT BLINKS IF THE ECHO FREQUENCY FROM THE IAPS DOES NOT MATCH THE TUNE FREQUENCY STORED IN RTU MEMORY.

Figure 16-18. RTU Radio Tuning Unit Controls

RTU FUNCTION KEYS

There are six function keys at the bottom of each RTU.

ATC ID

- ***ATC ID***—This key is active only on the RTU that is controlling the selected transponder. Depress this button to squawk ident. Ident can also be activated by depressing the IDENT button on either control wheel. ID replaces ATC, on the ATC display line, when ident is activated. The ATC REPLY annunciator at the top of the RTU illuminates when the transponder replies to an interrogation.

ATC STBY

- ***ATC STBY***—This button is active only on the RTU that is controlling the active transponder. Press this button to select the standby or the normal (on) mode.

DME HLD

- ***DME HLD***—Press the DME HLD button to hold the DME on the current frequency, and allow the NAV receiver to be independently retuned. When this button is pressed, the DME HOLD annunciator (located on top center of the RTU) illuminates and an H replaces NM following the DME distance on the PFD. The DME hold frequency is displayed on the NAV mode page. Press the DME HLD button a second time to cancel hold; the DME then retunes to the active NAV frequency; an H will also replace the NM following the DME distance on the SDU if in the VOR/LOC mode.

1/2

- ***1/2***—Normally each RTU controls only the on-side COM/NAV/ADF radios. Pressing and holding the 1/2 button will display the cross-side frequencies; however, these frequencies can only be changed when the cross-side RTU is disabled. Either RTU can be used to control both the left (1) and right (2) radios, but only when the cross-side RTU/OFF switch on the ECP has been selected to OFF (disabled). Pressing the 1/2 switch at the bottom of the RTU then selects either the left or right bank of radios, to be controlled by the RTU. The COM, NAV, ADF and ATC display lines show the selected side (1 or 2) next to the line keys.

COM SQ

- ***COM SQ***—This button can be depressed and held to, momentarily, disable the COM receiver squelch circuits. Squelch can also be turned on and off on the COM page.

ADF TEST

- ***ADF TEST***—Depressing this button verifies ADF signal strength. While the button is depressed and the signal strength is usable for navigation, the bearing pointer will swing 90° and a 1 kHz tone will be emitted.

RTU MODE CONTROL PAGES

The COM, NAV, ADF, and ATC mode pages can be selected by depressing the line key to move the selector star to the desired line and then depressing the same key a second time. Figure 16-19 shows an illustration of the mode pages. The RTU display will return to the main menu after approximately 15 seconds of inactivity on a mode page, or the RTN line can be selected on the mode page to immediately return to the main menu.

The ADF mode page allows antenna (ANT), or loop antenna (ADF), to be selected or deselected and the BFO (beat frequency oscillator) to be selected or deselected. The ADF receiver provides a 1,000 Hz aural output tone in BFO mode, when a keyed CW signal is received. The ATC mode page allows altitude reporting to be turned on or off and presents an uncorrected barometric altitude readout. The third line of the ATC page allows TCAS selection (if installed) to STBY, TA/RA, or TA. The selected mode annunciates by the line key. In the STBY mode the TCAS does not transmit. In the TA/RA mode, both traffic and resolution advisories may display on the PFD/MFDs. In the TA mode, traffic advisories (only) may display. The forth line on the ATC mode page allows transponder/TCAS to be tested by depressing the TST line key.

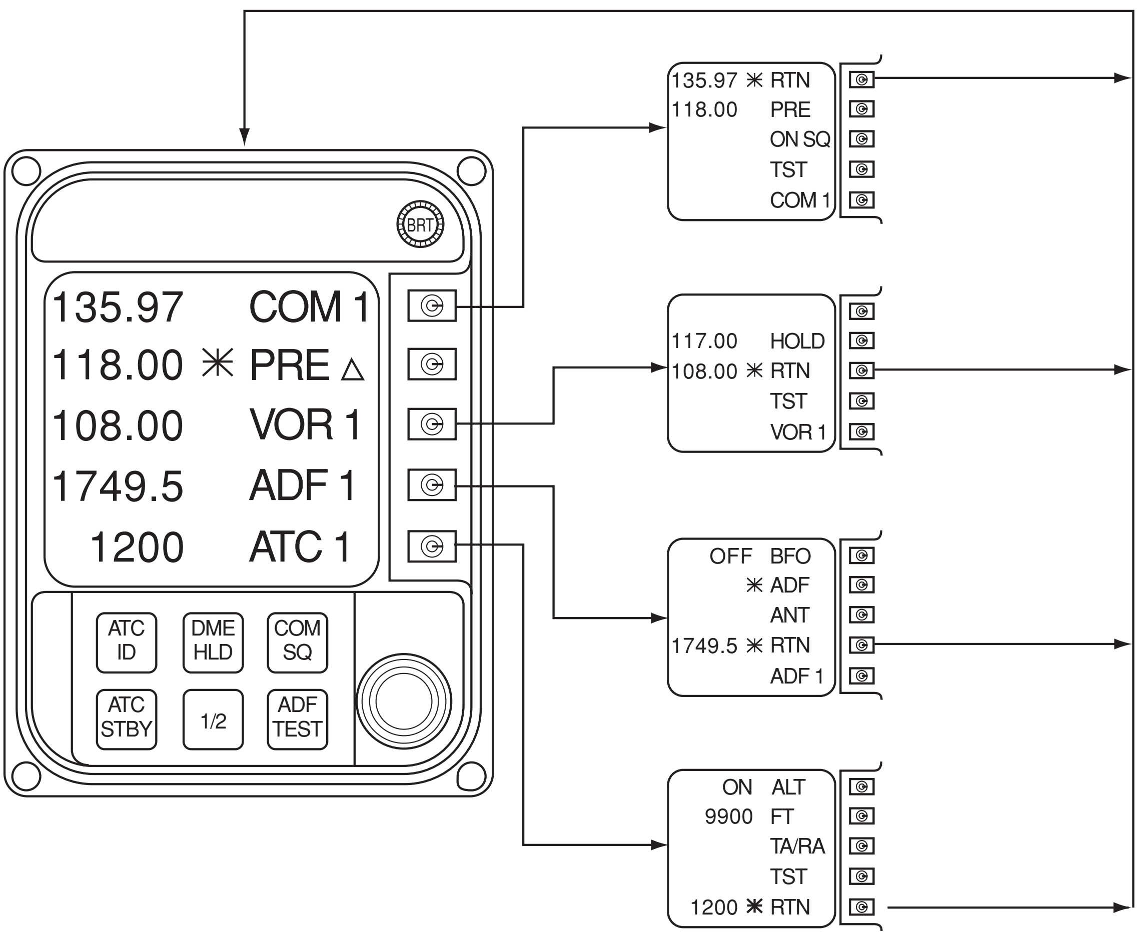


Figure 16-19. RTU Mode Control Pages

RADIO TUNING WITH THE UNS CDU

GENERAL

On aircraft equipped with the dual UNS-FMSs, the radio receivers and transponders can be tuned using the RTUs, as previously described, or through either UNS CDU on the center pedestal.

UNS-FMS CDU TUNE OPERATION

The TUNE page is used to tune the aircraft radios, select and store preselected frequencies for each radio, and to view the selected frequencies (active and preset) for each radio. The TUNE page is accessed from any mode by pressing the TUNE function key on the CDU (Figure 16-20).

When first accessed, the page will initially display the last tuned radio. The selectable radios are displayed next to the line select keys and a control window is located in the center of the display. Any applicable radio may be placed into the control window by pressing the line key for that radio.

The control window shows the selected radio, the active frequency, and up to four preset frequencies. When the TUNE page is first accessed, the control window will contain the active and preset frequencies of the last radio tuned and the cursor will be on the active frequency field. A new frequency may be entered into the active frequency, either directly, through the numeric keys on the CDU, or by inputting the reference number (1–4) of one of the preset frequencies. The frequency input is completed by pressing the ENTER key on the CDU.

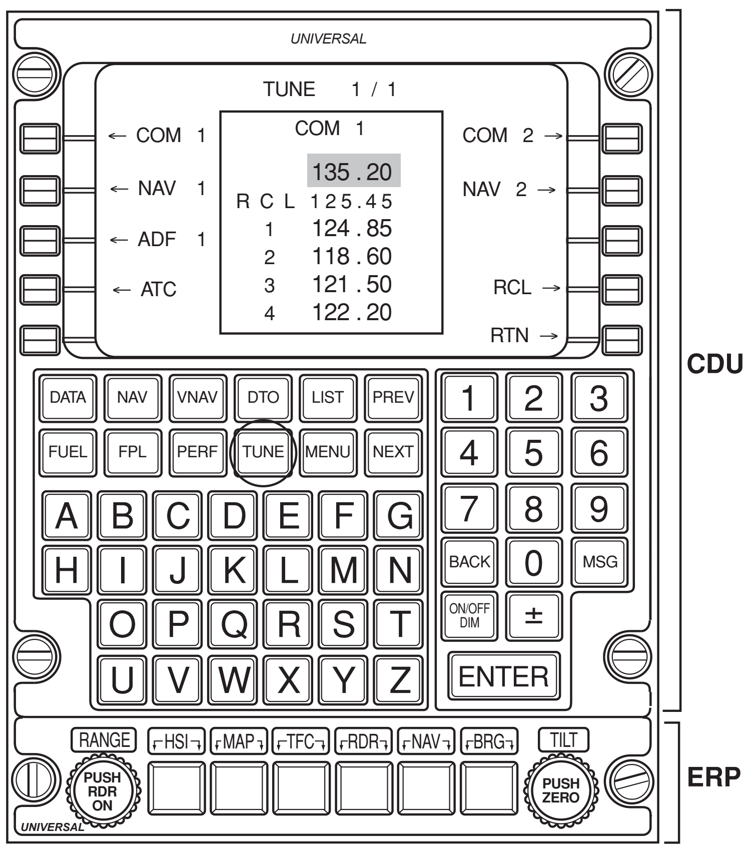


Figure 16-20. UNS-FMS CDU and TUNING Page

The TUNE page has a recall (RCL) line, below the active frequency, which will always contain the last frequency used whenever a new frequency is entered into the Active window. The RCL line key swaps the RCL frequency with the current ACTIVE frequency. Each of the four preset frequencies remain intact until modified by entering a different frequency.

Cursor movement in the control window is controlled using the ENTER key. If the ENTER key is pressed and nothing new has been entered into the active frequency, the cursor will advance to preset. Each time the ENTER key is pressed without entering a frequency first, the cursor advances to the next preset frequency, and after preset 4, it will return to the active frequency line.

The preset frequencies are changed or entered by placing the cursor over the desired preset position, inputting the new frequency, and then pressing the ENTER key. The new frequency may be inputted directly through the numeric keys or by inputting the reference number (1–4) of one of the preset frequencies. When typing in the frequency, the decimal point will be automatically placed in the proper position. For example, if a frequency of 135.65 is desired, input the numbers 13565. If 118.00 is desired, the number can be input as 118, 1180 or 11800. In either case, when the ENTER key is pressed, the display will read 118.00.

When tuning VOR and ADF frequencies, the operator has the option of either entering the actual frequency of the NAV aid, using the LIST function or direct input of identifier. The LIST key on the CDU is used to provide a list of options appropriate to the data to be entered. While performing data entry, pressing the LIST key presents a list of selections appropriate to the entry being made.

The ATC line select key on the TUNE page is used to gain access to control of the transponder codes. Preselects 3 and 4 are prefilled with codes 1200 and 7700, but can be changed if desired.

INSTRUMENT DISPLAY SYSTEM (IDS)

GENERAL

The primary units of the IDS are the four 7- X 6-inch electronic flight displays. Each has a pair of electronic instruments that integrate most display functions.

Primary data (attitude, airspeed, and altitude) is sent to the four large displays via dedicated ARINC 429 digital buses. All of the large displays have access to AHS, air data, and radio sensor data originating on both sides of the cockpit.

AVIONICS COOLING

The PFD and MFD contain internal fans to provide cooling. If a fan should experience a fault, the internal temperature monitor will detect the approaching thermal cutoff and the affected unit will display a red boxed DISPLAY TEMP which will flash for ten seconds before becoming steady. To prolong the display, the sky/ground raster (blue/brown background) will be removed from the attitude display. All other symbols on the display will remain functional. If the condition continues, the display will shutdown. As the display cools down, it will return. The PFD display can be turned off to prevent overheating/damage by selecting PFD REV on the EFIS control panel (see Figure 16-3)

There are two cooling fans under the glareshield (one on each side) which pull air from behind the instrument panel and exhausts it above the glareshield at the base of the windshield. Another independent fan circulates air from beneath the cockpit floor to the area behind the instrument panel to cool the CRTs. These fans are powered through the avionic fans circuit breaker on the copilot circuit-breaker panel. These equipment cooling fans are further described in Chapter 11—"Air Conditioning."

DISPLAY BRIGHTNESS CONTROL

The brightness of the PFD and MFD screens is controlled by individual brightness (BRT) knobs in the upper-left corner of the units and also by the EFIS dimmer knobs on the L and R INSTR LIGHTS panels.

PRIMARY FLIGHT DISPLAY (PFD)

GENERAL

The upper portion of the PFD provides the pilot with pitch, roll, and yaw information, flight director steering commands, flight director mode annunciation and vertical deviation. To the left of the attitude display is the IAS/Mach scale and associated references. To the right of the attitude display is the altitude scales (pressure altitude and radio altitude) and associated references. Directly below the attitude display is the HSI display and associated navigation information. To the left of the HSI is space for a vertical row of annunciators and to the right is the vertical speed indicator.

Attitude and heading information can be fed to the PFD by either AHS which is selectable on the left and right EFIS control panels. Normally the on-side AHS will be selected, but should it fail, the cross-side AHS can be selected. If AHS input data fails, boxed red ATT and MAG flags will be displayed and attitude and heading information is removed from the display.

Air data information can be fed to the PFD by either ADC and is selectable on the left and right EFIS control panels. Normally the on-side ADC will be selected, but should an ADC fail, the cross-side ADC can be selected. ADC failure would be evident with flags on the airspeed, altitude and vertical speed displays, and the loss of SAT temperature.

If the display tube for either PFD fails, the entire display can be moved to the MFD next to it by selecting PFD REV on the associated ECP (see Figure 16-3).

PFD DESCRIPTION/FUNCTION

Attitude Display

Refer to Figure 16-21 (Sheet 1) with the following description of the attitude section of the PFD. In the interest of brevity, the information that should be apparent from studying the illustrations in Figure 16-21 (Sheet 6) will not be described herein.

One thing not apparent from the drawings is that, above +30° or below –20° pitch, the PFD display is decluttered leaving only essential items. Chevrons come into view, directing the pilot to straight and level flight. Decluttering also occurs at bank angles greater than 60˚.

AP Annunciator

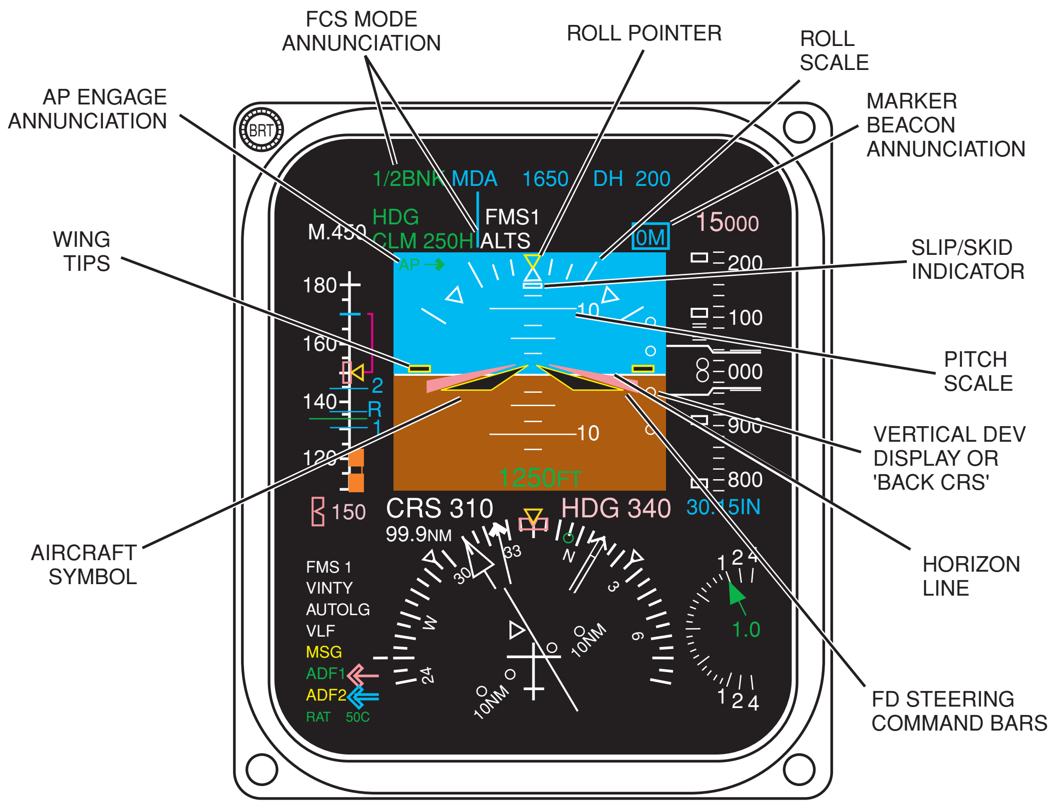
An autopilot engaged annunciator appears in the top-left corner of the attitude displays whenever the autopilot is engaged. The annunciator is a green AP► or AP◄, with the horizontal arrow pointing to the coupled side. When the autopilot disengages, this annunciator flashes amber for five seconds or until the autopilot disengage button is depressed. See the Autopilot/Flight Director section of this chapter for more information on the PFD autopilot annunciator.

FCS Mode Annunciation

Flight control system modes annunciate above the upper-left corner of the attitude display. Lateral modes display above vertical modes. Active modes display in green, and annunciate to the left of a blue vertical divider line. Armed modes display in white, and annunciate to the right of the line. FD modes are described in the Autopilot/Flight Director section of this chapter.

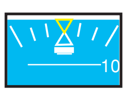
Flight Director Command Bars

Flight director (FD) V-bars appear in the center of the attitude display whenever a flight guidance mode is selected on the on-side flight control panel (FCP).



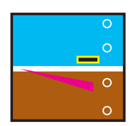
ATTITUDE

Figure 16-21. PFD Displays (Sheet 1 of 6)



Slip/Skid Indicator

A slip/skid indicator is located beneath the roll pointer. It moves laterally from the pointer to show aircraft slip/skid. This indicator is driven by lateral accelerations sensed by the AHS and replaces a traditional glass tube inclinometer.



Vertical Deviation

A vertical deviation display will appear on the right side of the attitude display when VNAV is selected, or when a ILS/MLS frequency is tuned, and is selected as the active navigation source. When making a back course approach, this display is replaced with a BACK CRS annunciator. If deviation data becomes invalid, this display is replaced with a red GS, MGP or VNV annunciator. The scale is conventional for ILS and MLS presentation and, for VNAV, each dot represents 100 feet of deviation.

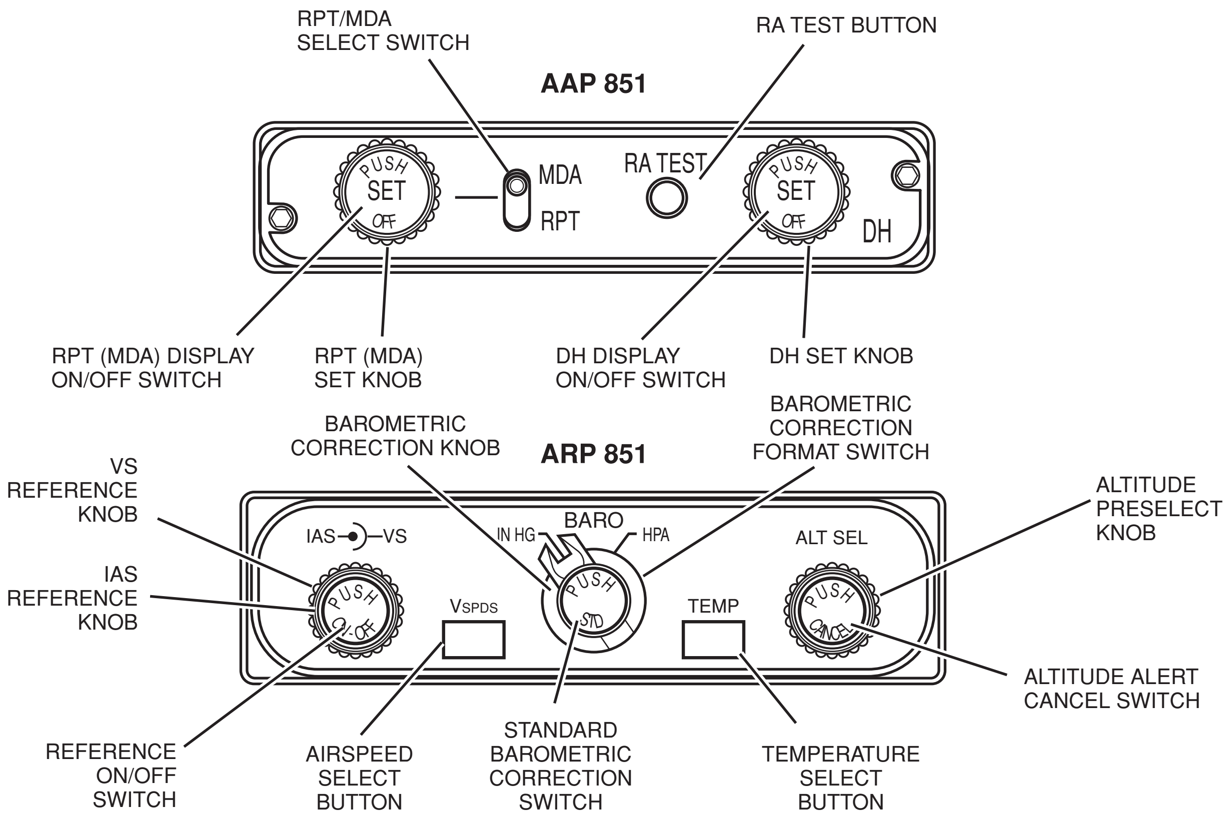
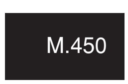


Figure 16-21. PFD Displays (Sheet 2 of 6)

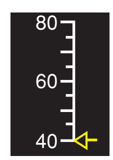
Airspeed Display

Refer to Figure 16-21 (Sheet 3) with the following description of the airspeed section of the PFD.



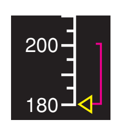
Mach Display

Mach is displayed as speed increases above .450M_I and the display is removed when speed decreases below .400M_I. The speed value is replaced with red dashes if Mach speed data becomes invalid.



IAS Scale

The indicated airspeed scale is a vertical “moving tape” display. If airspeed becomes invalid, the scale is replaced with a red boxed IAS annunciation.



IAS Pointer

This stationary, yellow triangle displays at the center of the IAS window. The current indicated airspeed is the IAS scale indication at the pointer.

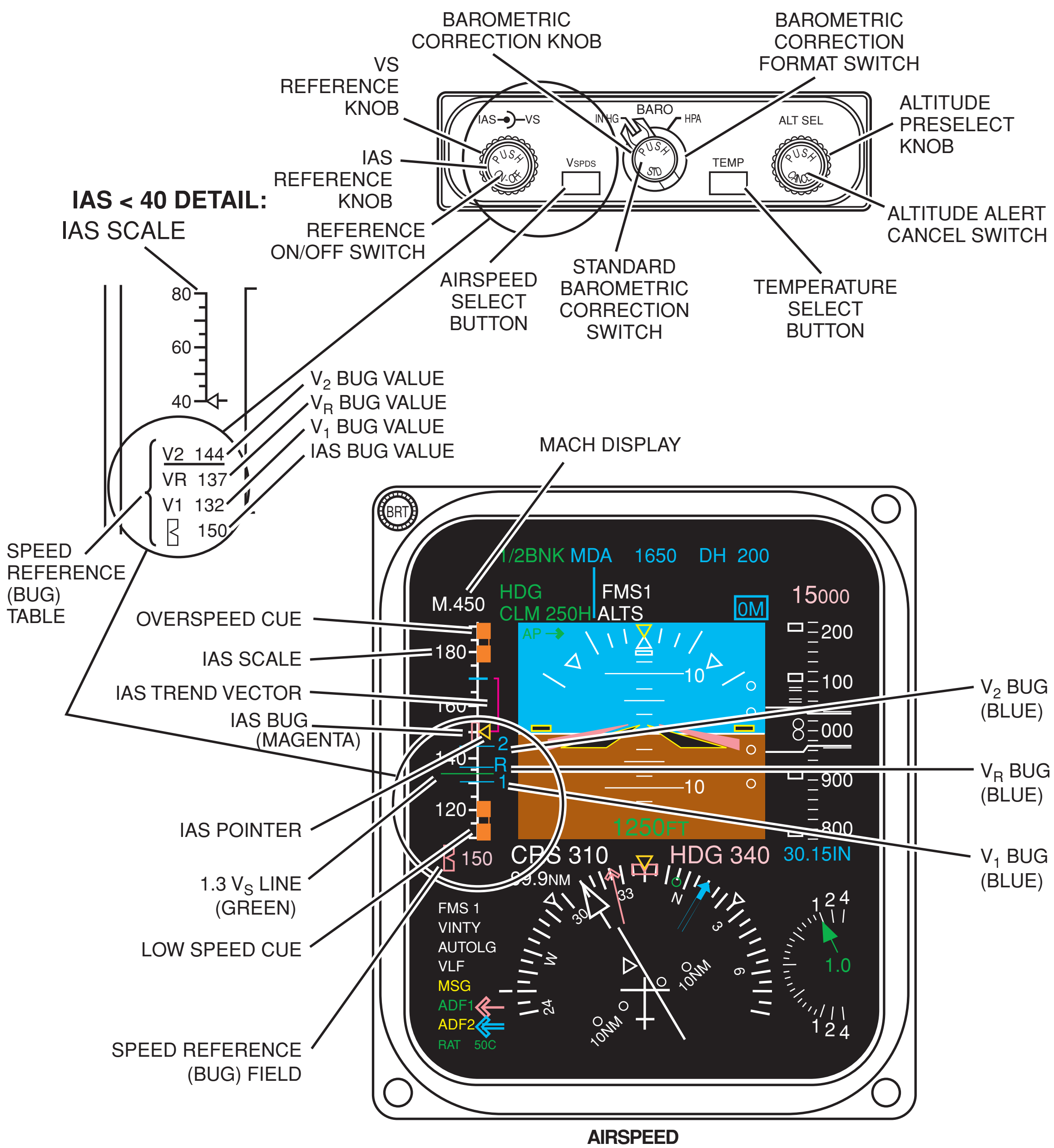
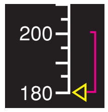
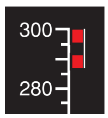


Figure 16-21. PFD Displays (Sheet 3 of 6)



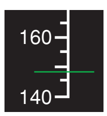
IAS Trend Vector

The IAS trend vector is a magenta line that extends from the IAS pointer, to predict future airspeed in ten seconds, based on present acceleration/deceleration. The trend vector is not displayed when the aircraft is on the ground.



Overspeed Cue

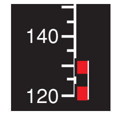
This red and black checkered bar moves up or down the IAS scale to show the maximum operating airspeed for current flight conditions. This value (V_{MO}/M_{MO}) is provided by the air data computer. The overspeed horn sounds if V_{MO} /M_{MO} is exceeded by 2 knots.



1.3V_S Line

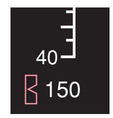
The green 1.3 V_S line and the low-speed cue display on the airspeed scale to show impending stall speed. These indications do not replace the stall warning system but provide a visual indication of approaching stall conditions. Both displays are calculated using angle of attack data. The green 1.3V_S line shows 1.3 times the computed stall speed. The 1.3V_S approach cue has a long filter time constant, to reduce excessive movement of the cue. The pilot should note that it may take up to one minute for the cue to reach its final value.

If the AOA system fails, the 1.3V_S line is replaced with a yellow bar that extends vertically from the low-speed cue. The top of the yellow default bar represents the highest stall speed of the aircraft (i.e., flaps up and heavy gross weight).



Low-Speed Cue

This red and black checkered bar rises from the bottom of the IAS window to the computed shaker speed. If the AOA system fails, the top of the low-speed bar represents the lowest stall speed of the aircraft (i.e. light weight aircraft with flaps down). The low-speed cue does not display on the ground.



Airspeed Reference Bugs

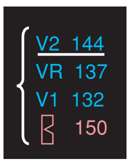
There are four airspeed references that can be set on the airspeed scale. Three of them (V_2, V_R, V_1) are blue lines that extend across the airspeed scale at the set value. These lines are identified with a 2, R, or 1 at the right end of the line. The forth reference is an IAS bug (magenta notched-box) that appears on the IAS scale at the set value. The four airspeed reference values are displayed in a speed reference (bug) table, at the bottom of the IAS scale, when on the ground, and the values can be displayed, one at a time, in the speed reference (bug) field, below the IAS scale, when in flight. V_1 and V_R reference lines are automatically removed from the airspeed scale five seconds after becoming airborne, and the V_2 reference line is removed when airspeed exceeds V_2 + 45 knots. Any of these reference lines can be reset in flight after being cancelled.

The airspeed reference values can be changed in the speed reference table on the ground, and in the speed reference field in flight, using the ARP. Setting V_1, V_R, or V_2 on one PFD automatically sets them on the other PFD; however, the IAS bugs have to be individually set on each PFD.

The IAS bug may be set to target speed or V_{REF} for landing or as desired for an IAS reference in flight. The other three speed references are

primarily for takeoff references, but can be used in flight if desired. For example, you may prefer to use the V_R blue line as a V_{REF} reference instead of the IAS bug.

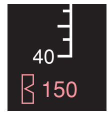
Any or all of the V_1, V_R, and V_2 references can be removed from the airspeed scale, if desired, but the IAS bug cannot. It can be set to a very low value, so that it does not appear within the normal airspeed scale range if you don't want it in view. Note that when the IAS bug reference is controlled by the FCS system (SPD or LVL CHG selected on the FD), the IAS value does not display in the speed reference field, but appears in the FCS vertical capture window of the PFD. When SPD is selected on the flight director, the speed can be changed by rotating the IAS reference knob on the ARP.



Speed Reference (Bug) Table

This table displays on the lower portion of the IAS scale when airspeed is less than 40 knots. When this table is displayed (aircraft on ground), press the V_{SPDS} button on the ARP to, sequentially, select one of the four available bugs. The selected bug and its numerical value will be underlined. The underlined bug value can then be changed by rotating the IAS reference knob on the ARP. The bug position on the airspeed scale updates on both PFDs when the value is changed (except IAS bug). The IAS bugs have to be set separately on the PFDs. The underlined bug (except IAS bug) can also be enabled/disabled for display on the airspeed scale by depressing the reference ON/OFF switch.

When electrical power is applied to the aircraft and the avionics master switches are turned on, a default value of 140 KIAS will appear for V_1, V_R and V_2 and a default value of 40 KIAS will appear for the IAS bug. Set values are retained if the avionics master switches are turned off, but if aircraft electrical power is turned off, they will revert to the default values.

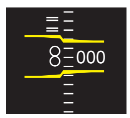


Speed Reference (Bug) Field

This one line display shows a selected reference (bug) value. Press the V_{SPDS} button on the on-side ARP (see Figure 16-21, Sheet 3) to, sequentially, select one of the four available bugs values for display in this field and then rotate the IAS reference knob, on the ARP, to change the value of that bug. The bug positions on both PFDs automatically update (only on-side IAS bug updates). The reference ON/OFF switch can be depressed to enable/disable display of that airspeed bug (except the IAS bug). After 15 seconds of inactivity in the reference field, the IAS bug display automatically selects.

Altitude Display

Refer to Figure 16-21 (Sheet 4) with the following description of the altitude section of the PFD.



Barometric Altitude Display

This display simulates a rolling drum mechanism and is outlined by a yellow window. The present barometric corrected altitude is the summation of the numeric "thousands" readout and the "hundreds" moving tape indication, at the window. If barometric altitude data becomes invalid, the altitude displays are replaced with a red boxed ALT annunciator.

Fine Baro Altitude Scale

This scale is a vertical "moving tape" display. The display window is 450 feet. The scale contains a line marking every 20 feet, and numeric label every 100 feet. The scale moves down for increasing altitude.

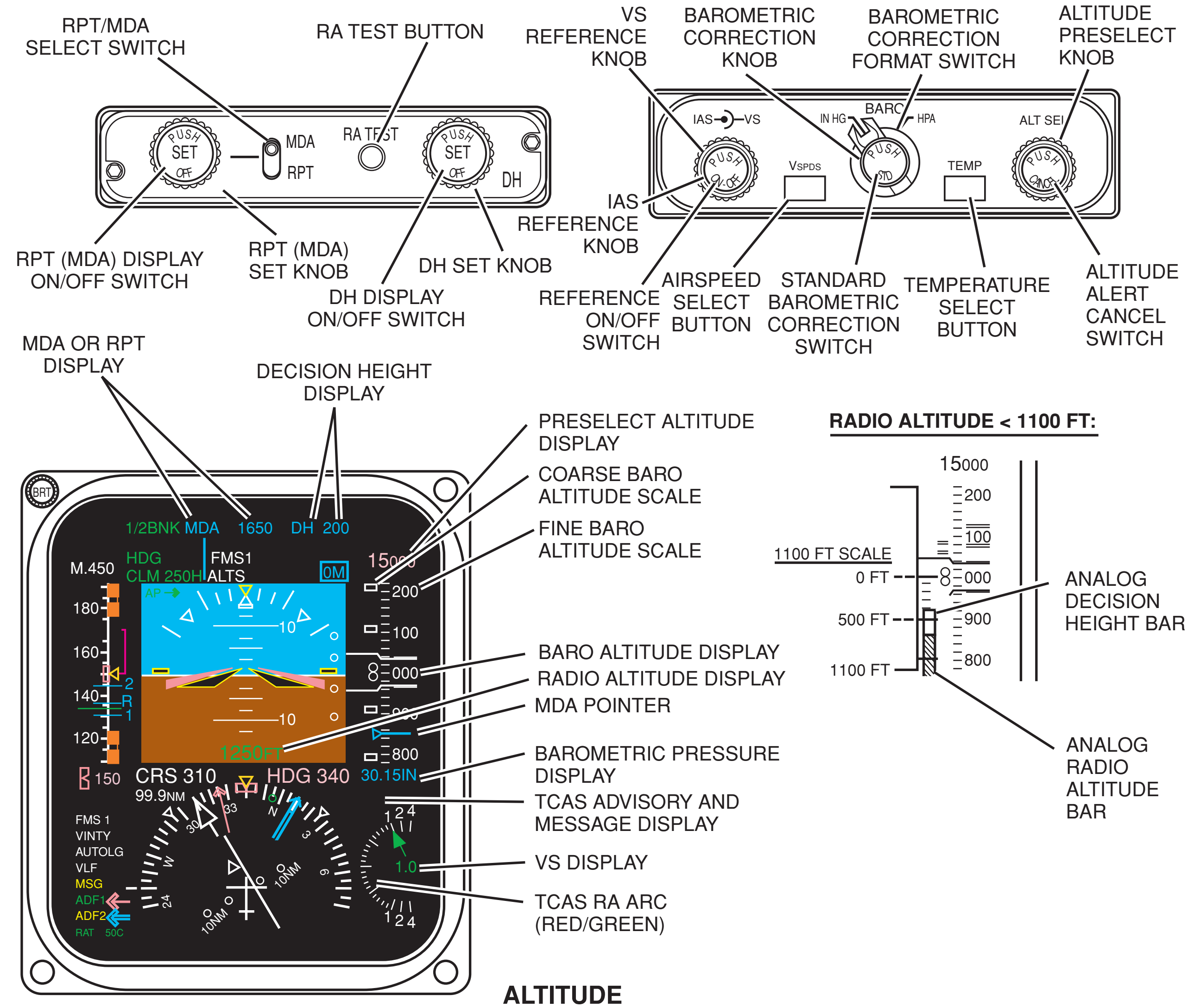


Figure 16-21. PFD Displays (Sheet 4 of 6)

Coarse Baro Altitude Scale

This scale is a non-numbered, vertical "moving tape" display that helps visualize approaching preselect altitudes. Large rectangles on the scale represent 1,000 foot altitude increments; small rectangles represent 500 foot increments. The display window is 2,200 feet. The scale moves down for increasing altitude.

Barometric Pressure Display

Barometric pressure correction (altimeter setting) is numerically displayed in inches of mercury (IN), or in hecto pascals (HPA), at the bottom of the altitude scale. The format (IN or HPA) and the altimeter setting are set with the format switch and the baro correction knob at the center of the on-side ARP (Figure 16-21, Sheet 4). The altimeter settings have to

be individually set on each PFD. Standard barometric correction (29.92 or 1013) can be set by depressing the standard barometric correction switch on the on-side ARP. The numerical display flashes when passing through 18,000 feet, or FL 180, as a reminder to reset the altimeter.

Preselect Altitude Display

The preselect altitude numerically displays magenta above the baro altitude scales. This value is simultaneously set on both PFDs with the altitude preselect knob on either ARP (Figure 16-21, Sheet 4).

Preselect Altitude Bug

This 4-line bug (double magenta lines above and below the selected altitude) displays on the coarse, and/or fine baro altitude scales, to mark the preselect altitude value. This value is numerically repeated in the preselect altitude display above the altitude scale.

A tone sounds, and the preselected altitude display and bug (4-line marker) both flash when the aircraft closes to within 1,000 feet of the preselected altitude. When the aircraft closes to within 200 feet of the preselected altitude, the preselected altitude display and the bug become steady. After capture, the bug (only) flashes magenta for minor altitude deviations (100 feet). The bug, and the numeric preselected altitude display, both flash yellow for major altitude deviations (200 feet) and a tone sounds. The flashing alert and the tone can be canceled by depressing the cancel in the altitude preselect knob.

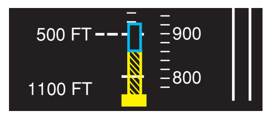
The displays become blue if the left and right altitude preselect values do not track with each other. They can be resync'd by setting the preselect value displayed on the cross-side PFD.

Radio Altitude Display

This green numerical display appears at the bottom center of the attitude display as the aircraft descends through 2,500 feet. If radio altitude data becomes invalid, this display is replaced with a red RA annunciation.

Radio Altitude Test

When the RA TEST button on the AAP is pressed, the radio altimeter is placed in self-test mode and outputs a fixed value of 50 feet. If the decision height is set higher than the self-test value, the DH annunciator turns on during the self-test.



Analog Radio Altitude Bar

An analog radio altitude pictorial displays as the aircraft descends through 1,100 feet. A yellow (ground reference) bar appears with an 1,100 foot radio altitude scale. At 0 foot radio altitude, the top of the ground reference bar will have risen to align with the center of the baro altitude window.

Decision Height Display

Selected decision height numerically displays with a DH label above the top-right side of the attitude display. The DH display is selected ON or OFF by depresssing the center of the right set knob of the AAP (Figure 16-21, Sheet 4). The DH displays must be turned ON and OFF individually on each side, but the DH value simultaneously sets on both PFDs with rotation of the DH set knob on either AAP. The decision height range is 0 to 999 feet settable to one foot increments. This value is replaced with red dashes, if decision height data becomes invalid.

A yellow DH annunciates near the center, right side, of the attitude display when the aircraft is at or below the DH value. It flashes for approximately 10 seconds when the set DH altitude is reached and then goes steady.

Analog Decision Height Bar

Decision height also displays on the analog radio altitude pictorial. A blue bar extends from the top of the yellow (ground reference) bar to show decision height. The DH value is the bar length, and is read against the 1,100 foot scale.

MDA or RPT Display

Either minimum descent altitude or reporting altitude can be numerically displayed, above the attitude display, at the center of the screen. MDA or RPT may be chosen as an altitude reference, with a two-position toggle switch, near the center of the AAP (see Figure 16-21, Sheet 4). The display is selected ON or OFF by depressing the center of the left set knob on the on-side AAP. The altitude (MDA or RPT) is simultaneously set on both PFDs by rotating the set knob on either ARP, but the display of MDA/RPT must be turned ON and OFF individually on each side.

The range of MDA is 0 to 12,000 feet with a 10 foot readout accuracy and the range of RPT is 0 to 60,000 feet.

MDA 7850 DH 100

A yellow MDA alert displays near the center, left side of the attitude display when the aircraft is at or below the minimum descent altitude. The yellow MDA flashes for approximately 10 seconds when the set MDA altitude is reached and then goes steady.

RPT 7850 DH 100

A RPT alert displays when the aircraft is at the reporting altitude. The numerical RPT display flashes and then becomes steady.

MDA Pointer

A blue pointer marks the selected minimum decision altitude on the fine baro altitude scale. This pointer displays when MDA is selected on the AAP, and flashes when the aircraft descends below the pointer.

Vertical Speed Display (VS)

VS Display

Current vertical speed is shown by a pointer on a semi-circular scale in the lower-right corner of the PFD, and numerically repeated in the center of the scale. If vertical speed data becomes invalid, this display is replaced with a red boxed V/S annunciation.

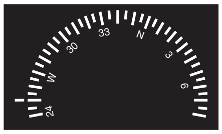
The white VS scale contains nonlinear markings to increment vertical speeds between –4,000 and +4,000 feet per minute. A pointer rotates along the scale to show the current vertical speed value.

The VS value is numerically repeated in the center of the scale up to a value of ±15 (15,000 FPM). Above 9,950 fpm, the digital value is rounded to the nearest 1,000 fpm. The pointer and numerical VS value are normally green, but become red if the current vertical speed is within a red TCAS advisory arc.

When VS is controlled by the FCS (VS or LVL CHG below 8,000 feet), the VS value display in the FD vertical capture window can be changed with the VS knob on the on-side ARP.

HSI Display

Refer to Figure 16-21 (Sheet 5) with the following description of the HSI section of the PFD.

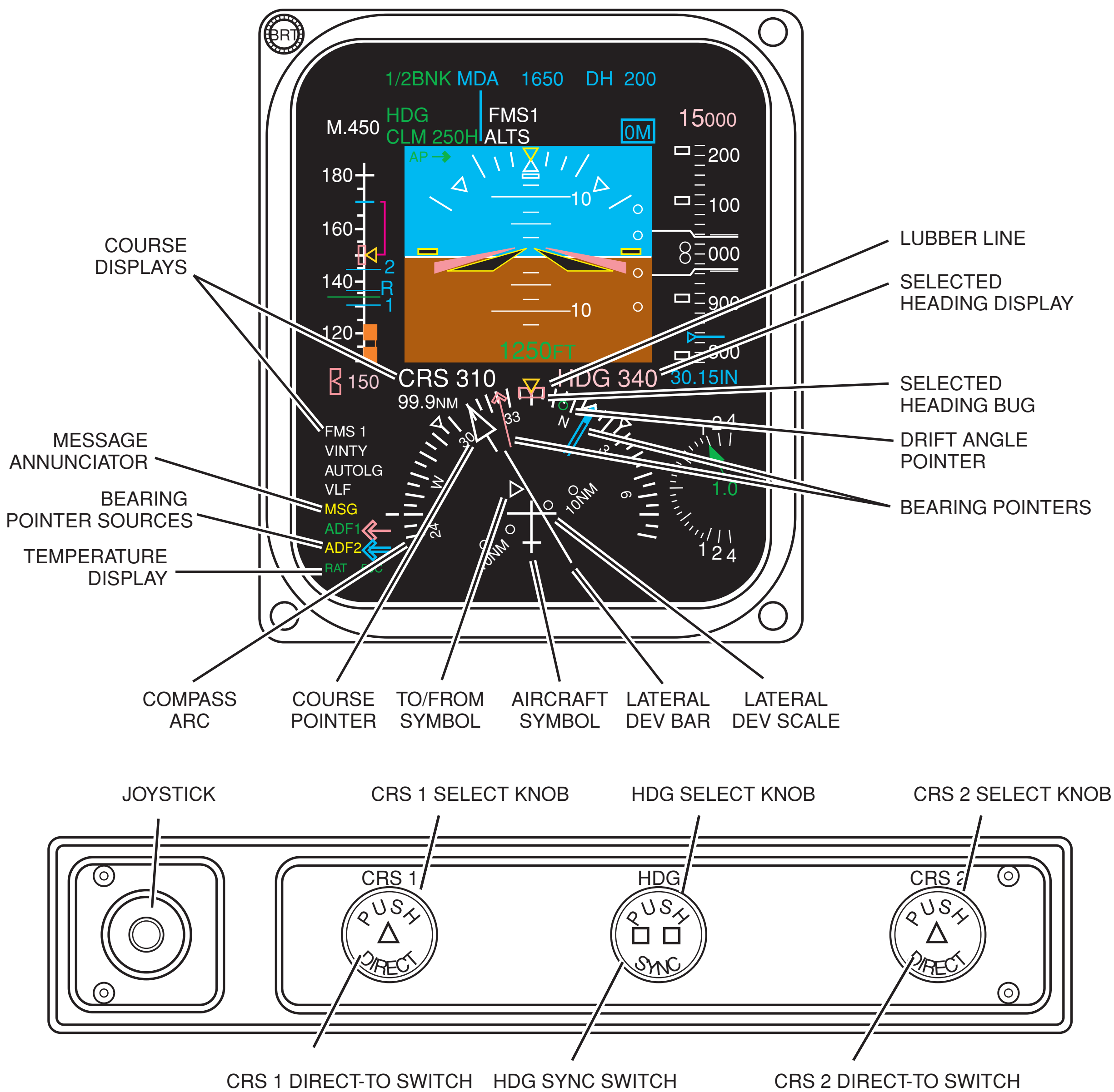


Compass Arc

A 200° section of the compass rose displays on the lower portion of the PFD. A full compass rose can be displayed on the MFD when the HSI format is selected on the CDU.

Heading Bug

A magenta heading bug symbol is provided on the compass arc to mark a selected heading reference. Turn the HDG knob on the CHP (Figure 16-21, Sheet 5) to move the heading bug around the compass arc. There is only one heading select knob on the CHP. The heading bugs on both PFDs and MFDs all change to the



HSI AND CHP

Figure 16-21. PFD Displays (Sheet 5 of 6)

FCS MODE ANNUNCIATION

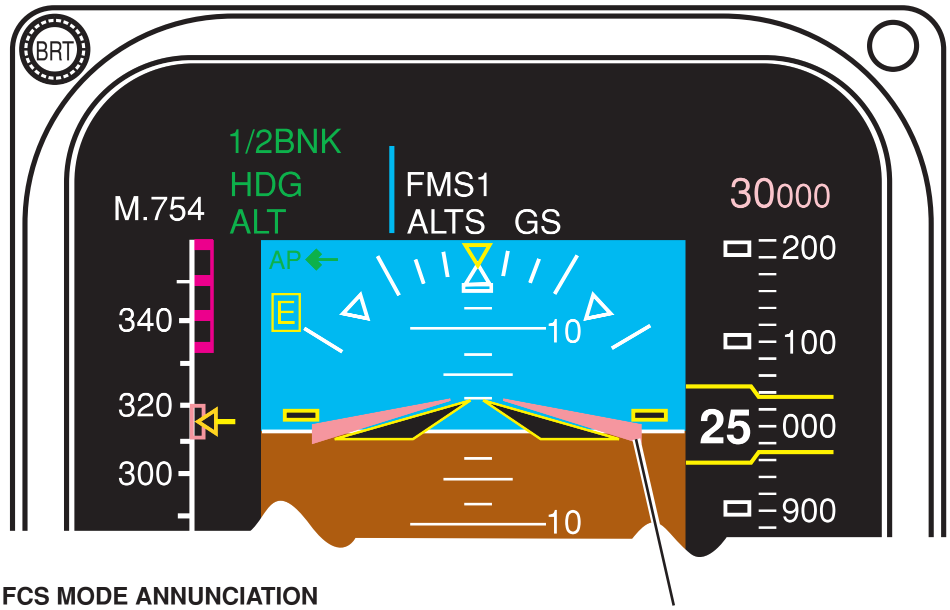
*Active/Captured Lateral Mode				Armed Lateral Mode	
ROLL	(G)	FMS1	(G)	VOR1	(W)
HDG	(G)	FMS2	(G)	VOR2	(W)
VOR1	(G)	GA	(G)	LOC1	(W)
VOR2	(G)	1/2 BNK	(G)	LOC2	(W)
LOC1	(G)			FMS1	(W)
LOC2	(G)			FMS2	(W)

AP ENGAGE ANNUNCIATION

AP ←	(G)
AP →	(G)
←	(W)
→	(W)
AP ←	(Y)
AP →	(Y)
YD	(Y)

MISTRIM ANNUNCIATION

E A R (Y)



FLIGHT DIRECTOR COMMAND BARS (V-BARS)

FCS MODE ANNUNCIATION

*Active/Capture Vertical Mode			
PTCH	(G)	DES XXXH	(G)
ALT	(G)	DES .XXH	(G)
ALTS	(G)	DES X.X↓	(G)
ALTS CAP	(G)	IAS XXX	(G)
CLM XXX	(G)	MACH .XX	(G)
CLM .XX	(G)	VS X.X↑	(G)
CLM XXXH	(G)	VS X.X↓	(G)
CLM .XXH	(G)	VNV	(G)
DES .XX	(G)	GS	(G)
DES XXX	(G)	GA	(G)

Armed Vertical Mode	
VNV	(W)
GS	(W)
ALTS	(W)
ALTS	(Y)

LEGEND

(G)	GREEN
(W)	WHITE
(Y)	YELLOW
.XX	MACH REFERENCE VALUE
XXX	IAS REFERENCE VALUE
X.X	V/S REFERENCE VALUE

*Displayed as ~~xxxxx~~ when mode becomes invalid (e.g., ~~VOR1~~).

Figure 16-21. PFD Displays (Sheet 6 of 6)

same heading with the heading select knob. A PUSH to SYNC switch is located in the center of the HDG select knob and will move the heading bug to the lubber line of the HSI.

The heading bugs also reposition when in the FMS HDG mode and a heading is entered using the CDU.

The heading bug on the SDU is set with a heading set knob on the SDU.

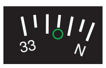
When the heading bug is positioned outside of the 200° arc on the PFD, a magenta dashed line extends from the aircraft symbol to provide an angular indication of the selected heading value.

NOTE

Do not select a heading change greater than 180°. This will cause the FD to reverse the direction of turn (turn in the shortest direction to the holding selected).

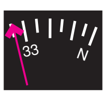
Heading Display

There is a magenta HDG and digital heading display above the compass rose to the right of the lubber line. This is a numerical readout of the heading bug position.



Drift Angle Pointer

This pointer is a small green circle that rotates left or right of the lubber line on the compass arc to show the aircraft drift angle. The amount of aircraft drift is the angular difference between the pointer position (circle) and the aircraft heading. When the pointer is under the lubber line, the drift angle is zero.



Bearing Pointers

Two bearing pointers can be selected for display on the PFD and MFD using the CDU and ERP. Selecting bearing pointers for display on the PFD and MFD is described in this chapter under UNS-FMS CDU and ERP Operation.

One pointer is a single bar, V-head pointer and reciprocal tail and the other pointer is a dual bar, V-head pointer and reciprocal tail. Each pointer shows the bearing to a selected navaid station or the next waypoint. Due to the partial compass rose display, you may only see the V-head pointer or reciprocal tail of the bearing pointers.

Bearing Pointer Sources

The NAV source(s) selected to drive the bearing pointer(s) are annunciated just to the left of the compass rose. One, both, or none of the bearing pointers may be selected for display. If the selected source fails, or is invalid, the source annunciation becomes boxed and turns red and the V-head pointer is removed from view.

Course Display

The digital course display is presented to the left of the compass arc lubber line. The display consists of CRS and a three digit course value.

The course value is calculated by the FMS or selected by the CRS knob on the CHP (see Figure 16-21, Sheet 5). The CRS 1 knob (left side of the CHP) controls the course on the left PFD and MFD and the CRS 2 knob (right side of CHP) controls the course on the right PFD and MFD. The SDU has a set knob on the bottom-right corner of the SDU which can be used to select a course in the VOR and LOC formats.

The course display has a distance readout beneath it. The NM readout shows the distance to the tuned navaid or next waypoint (in nautical miles).

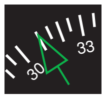
NOTE

An H replaces the NM label if in DME hold mode. The NM label is replaced by dashes or blanked if valid data is not available.

NAV Source

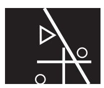
The selected active Nav source is displayed to the left of the compass arc and directly beneath that, the station or next waypoint. The active NAV source is selected using the CDU and ERP and is described in this chapter under UNS-FMS CDU and ERP Operation.

Annunciation on the forth line beneath the active NAV source identifier may include DR (if FMS is in dead reckoning) or GPS (if FMS is in GPS only mode). If the selected active NAV source fails or is invalid, the source annunciation becomes boxed and turns red.



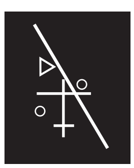
Course Pointer

The course pointer is a solid-line, triangle-head pointer with a straight line tail. This pointer shows the active NAV course, and is numerically repeated in the CRS display.



To/From Symbol

A triangle symbol shows "to" or "from" direction. This symbol rotates as a part of the course pointer, and points toward the tuned station or next waypoint.



Lateral Deviation Bar and Scale

This bar moves left or right from the course pointer head and tail to show lateral deviation from the active NAV course. The lateral deviation scale consists of four dots that display perpendicular to the lateral deviation bar. Two dots display on either side of the aircraft symbol. In FMS modes, distance labels display by the outer dots to show lateral deviation from course in nautical miles.

Temperature Display

This green temperature display is located at the bottom of the PFD screen on the left side. It shows SAT (static air temperature), RAT (ram air temperature), or ISA (deviation from international standard atmosphere temperature). While the display normally shows SAT, to select one of the other temperature formats, depress the TEMP button on the on-side ARP (Figure 16-21, Sheet 2). Press the button once to select RAT or press it twice to select ISA. After approximately five seconds, the display will revert to SAT. The temperature is repeated on the status line in the top-right corner of the MFD.

MSG Annunciator

MSG annunciates in yellow to the left of the compass rose when an unread system message exists in the on-side FMS. Press the MSG key on the CDU to review the system messages. Refer to the *UNS-FMS Operator' Manual* for further information on use of the MSG key.

PFD/MFD Annunciators

Table 16-4 shows a list of annunciators that may appear at various locations on the PFDs or MFDs.

PFD Electrical Power Source

The left and right PFDs receive DC electrical power through the PFD 1 and PFD 2 circuit breakers, on the left and right circuit-breaker panels, respectively. They are not powered during the emergency bus mode of operation.

Table 16-4. PFD/MFD ANNUNCIATORS

MESSAGE	COLOR	MEANING
ADC # (1 or 2)	yellow	Cross-side air data source is selected.
ALT	red	Altitude data is invalid.
ATT # (1 or 2)	yellow	Cross-side AHS is selected.
ATT (boxed)	red	Attitude (AHS) failure is detected.
ATT/HDG ALIGNING	white	The AHS is initializing.
CDU (boxed)	yellow	CDU failure is detected.
CDU # (1 or 2)	yellow	Cross-side CDU/ERP is selected.
DH	yellow	Decision height alert.
DISPLAY TEMP	red	PFD/ND/MFD is overheating.
E, A or R (boxed)	yellow	Elevator, Aileron or Rudder mistrimmed.
FD (boxed)	red	Flight Director failure is detected.
GS	red	Glideslope deviation data is invalid.
GS (boxed)	yellow	Comparator detects GS dev disagree (Cat II only).
HDG (boxed)	yellow	Comparator detects hdg disagreement.
IAS	red	Airspeed data is invalid.
LOC (boxed)	yellow	Comparator detects loc dev disagree (Cat II only).
MAG (boxed)	red	Mag Heading (AHS) failure is detected.
MAG # (1 or 2)	yellow	Cross-side AHS is selected.
MDA	yellow	Minimum decision altitude alert.
MSG		System message (yellow/white CDU).
NO FLIGHT PLAN	white	No flight plan is entered in FMS.
PIT (boxed)	yellow	Comparator detects AHS pitch disagreement.
RDR CTL FAULT		WXR/CDU Range Discrepancy is detected.
RDR FAULT		RTA Internal Failure or Failed CDU Input.
RA	red	Radio altitude data is invalid.
RA (boxed)	yellow	Comparator detects radar alt disagree (Cat II only).
ROL (boxed)	yellow	Comparator detects AHS roll disagreement.
SPD LIM	red	One or more speed prompts invalid.
TA only		TCAS is in TA mode (white, yellow if TA intruder).
TCAS FAIL		TCAS system failure.
TCAS OFF		TCAS is in standby mode.
TCAS RA FAIL		PFD cannot display TCAS resolution advisory.
TCAS TEST		TCAS computer is in test mode.
TD FAIL		TCAS Data is invalid.
TEST		FCS is in test mode.
TRAFFIC	yellow/red	TA (yellow) or RA (red) intruder.
VNV	red	VNAV deviation is invalid.
V/S	red	Vertical speed data is invalid.
YD	white	Yaw Damper is disengaged.

UNS-FMS CDU/ERP OPERATION/ DESCRIPTION

NOTE

The Learjet 60 could have one of several different UNS-FMS models installed as options or original equipment. Basic operation of the units is the same.

GENERAL

When dual UNS FMSs are integrated with the Collins Proline 4 avionics package in the Learjet 60, they are individually controlled with dual Universal CDUs located on the forward part of the center console. Dual UNS installation also includes two EFIS radar panels (ERPs), mounted directly aft of the CDUs, to supplement the CDU controls. The CDUs/ERPs are the controls for the PFD and MFD displays, FMSs, VNAV, TCAS, and radar.

This section of Chapter 16 will describe the functions of the Universal CDUs and ERPs that control the EFIS displays (PFDs and MFDs).

The functions of the CDUs for long-range navigation (FMS), vertical navigation (VNAV), traffic collision avoidance system (TCAS) are not covered in this chapter. Refer to the *Universal UNS Operator's Manual* for information on these subjects.

The CDUs also provide an additional method (other than the RTUs) for changing radio frequencies and transponder codes. The CDU/ERP functions for radio tuning and radar control are also described in this chapter under Radio Tuning and Radar Operation.

The CDUs use a combination of display menus, line keys, a full alphanumeric keypad and dedicated control keys. The ERPs are integrated with the CDUs to provide some additional related control functions through a set of control knobs and a row of dedicated control keys.

One CDU controls one FMS, and the other CDU controls the other. They work independently and do not automatically interface or cross-feed except for radar channel (split/sync). However, there are three functions which can be crossfilled from one FMS to the other FMS: initialization, flight plan, and fuel. These functions can be crossfilled separately or as a group (MSTR XFILL) if so desired. MSTR XFILL is only available on the ground.

The CDUs/ERPs are also used independently to control the on-side PFDs and MFDs. Either CDU/ERP can control the radar and tune radios. There is no limitation to simultaneous operations.

If there is a malfunction in one of the CDU/ERPs, depressing the CDU cross-side switch on the failed-side EFIS control panel allows the cross-side CDU/ERP to control the failed side ERP functions. With the CDU cross-side switch selected, the failed-side MFD will follow the cross-side MFD display. The failed-side MFD line select keys will be inoperative, and only three flight plan waypoints will be displayed. The cross-side flight plan and navigation functions will be displayed on the failed-side MFD. The failed-side CDU function keys will not affect the displayed flight plan and navigation. The failed-side PFD navigation source (NAV) and bearing pointer selection (BRG) will also follow the cross-side PFD display. Course knob and AAP functions on the failed side will be inoperative. The operative course knob and AAP will control both flight displays. A CDU 1 will appear above the active NAV source annunciator on both PFDs if CDU cross-side is selected on the copilot side, and a CDU 2 will appear if cross-side is selected on the pilot side. The pilot's side cross-side selection will override the copilot's cross-side selection if both sides are selected at the same time.

DESCRIPTION/OPERATION

CDU Display Format

The CDUs (Figure 16-22) each have a display screen that has eleven lines, on which menus and data can be displayed in two character sizes, using colors in the red and green spectrum to differentiate the type of data being displayed. The standard color format is green characters on a black background. Background colors and/or boxes are used to group data into important areas such as control or selection boxes.

NOTE

Newer UNS models have an LCD screen with higher resolution (graphics capable). The color coding is different, but obvious. The hardware is significantly different, but does not affect basic operations.

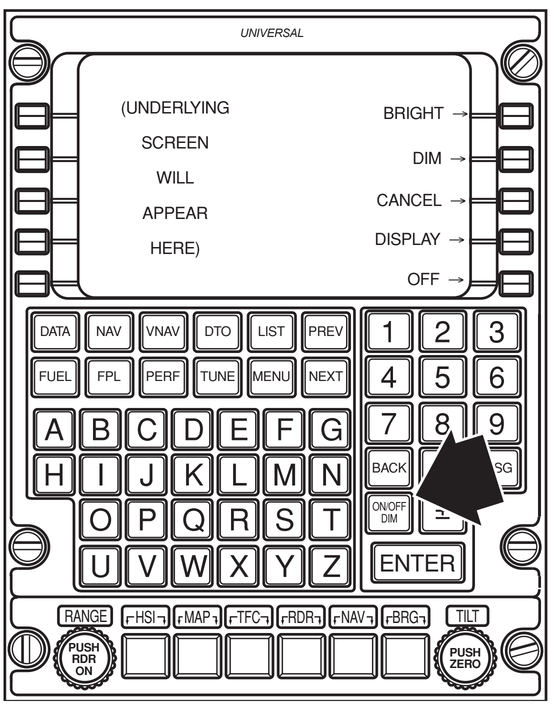


Figure 16-22. UNS FMS ON/OFF—DIM Key

Line Select Keys

There is a vertical row of five line select keys on each side of the display screens. These line select keys may be used to make menu selections or to position the cursor. When outward pointing arrows are present at the far left or right end of a display line, it indicates that the adjacent line select key is active for menu selection.

Data may be input at the cursor position with the alphanumeric keys; and then, the ENTER key is depressed to complete the entry. If there is a logical next field for data entry, the cursor will automatically advance to this next field when the ENTER key is pressed.

Selections are made with the line select keys whenever possible. In some cases a combination of line select keys and reference numbers are used on the same display page. This allows two levels of selection to exist simultaneously on the same display. For example, while the contents or nature of a list are controlled by the line select keys, an item from that list can be selected by using a reference number.

CONTROL KEYS

ON/OFF—Dim Key

The ON/OFF—DIM key (Figure 16-22) provides power-up, display dimming, display alignment, and unit shutdown functions. The FMSs can be turned on and programmed without turning the avionics masters on. See *AFM* for specific power options.

Pressing the ON/OFF—DIM key for initial power-up will energize the system and initiate self-test of the navigation computer. The self-test page will automatically be followed by the initialization page if all tests are successfully completed. If a failure that would cause the system to be unusable occurs, the initialization page will not appear. Once the initialization page appears, no other page can be displayed until the initialization data is accepted. After the system is turned on, pressing the ON/OFF—DIM key will cause a control window to be displayed on the right side of the active page, with the options BRIGHT, DIM, CANCEL, DISPLAY, and OFF selectable using the line select keys (Figure 16-22).

Bright

Pressing the line select key for BRIGHT will cause the display screen to steadily brighten as the ON/OFF—DIM key is held depressed.

Dim

Pressing the line select key for DIM will cause the display screen display to steadily dim as the ON/OFF—DIM key is held down.

Cancel

Pressing the line select key for CANCEL will cause the control window to be removed from the active display page.

Display

Pressing the line select key for DISPLAY will cause the parallax adjustment window to be displayed. The parallax adjustment window presents three options (UP, DOWN, and CANCEL), selectable using the line select keys. Selecting UP will cause the entire display to shift upwards by as much as one-half character to adjust the parallax for the line select keys. Selecting DOWN will adjust the display downwards an equal amount. Selecting CANCEL will return the display to the main (BRIGHT/DIM/CANCEL/DISPLAY/OFF) window.

Off

Pressing the line select key for OFF will cause the CONFIRM OFF window to be displayed. This window has two options (CONFIRM OFF and CANCEL), selectable using the line select keys. Selecting CONFIRM OFF will turn the system off. Selecting CANCEL will return the display to the main (BRIGHT/DIM/CANCEL/ DISPLAY/OFF) window.

PREV Key

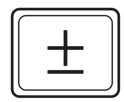
The PREV (previous) key is used to cycle backward, one page at a time, through multiple pages of the same mode.

NEXT Key

The NEXT key is used to cycle forward, one page at a time, through multiple pages of the same mode.

BACK Key

When the cursor is over a data entry field, the BACK key serves as a delete or backspace key.



± Key

The state change key (±) is used in conjunction with the alpha numeric keys to enter data. It changes + to –, N to S, and L to R. It is also used in strictly alpha fields as a dash or period.

MSG Key

When a system message becomes active, MSG will appear on the PFD and in orange on the far right side of the top line of the display screen. If the position uncertain message is active, POS will be displayed in orange on the far left side of the top line on the display screen.

Pressing the MSG key will cause the MESSAGE page to be displayed showing the active messages. The current messages (those messages generated since the page was last accessed) will be displayed in orange. After the messages are viewed, the display screen may be returned to the previous page by selecting the RETURN option on the MESSAGE page, by pressing the MSG key again, or by pressing the BACK key. Selecting the AFIS option will access the AFIS menu page.

ENTER

ENTER Key

The ENTER key is used to store input data. The cursor marks variable parameters by means of reverse field printing (dark letters on a light background). Parameters that cannot be marked by the cursor are not variable and cannot be changed by the normal input processes. Each time the ENTER key is pressed, the variable marked by the cursor will be stored in memory. When the cursor marks a variable, it may be altered through the alphanumeric keys and then stored by pressing the ENTER key. Pressing the ENTER key completes entry of data and is required for all data entries.

Function Keys

Eight function or mode select keys are located immediately below the display screen. These keys are used to select the basic operating modes of the system for data entry or command inputs. When one of the function keys is pressed, the display screen display will immediately change to the first display page of the selected mode. Where multiple pages exist, subsequently pressing the function key will cycle the display forward one page at a time. See the *UNS Operator's Manual* for a detailed description of each mode and the various display pages under each mode.

DATA

DATA Key

The DATA function is used to obtain information and status about the FMS, its NAV data base, and the attached sensors that operate with the FMS. Although sensor control is totally automatic, selection and deselection of individual sensors may be accomplished using this function. The DATA function is also used to make additions, deletions, or changes to pilot defined locations.

EFIS RADAR PANEL (ERP)

The EFIS radar panel (ERP) enables the FMS to function as an external navigator to the Collins Pro Line 4 avionics system. The ERP is used to control weather radar and TCAS. Two ERPs are installed with the dual UNS FMSs. The ERPs are installed directly aft of the FMS CDUs on the center pedestal console.

Range Knob and Tilt Knob

There are two control knobs on each ERP—one at each end of the panel. The knob on the left is used to control MFD radar and present position map display range (up to 300 NM range). It also controls the TCAS display range (up to 40 NM). There is a PUSH RDR ON switch in the center of the left knob to alternately turn the radar system and MFD radar display on and off.

The knob at the right end of the ERP controls radar antenna tilt. It has a push type switch in the center of the knob labeled PUSH ZERO. It can be used to select zero degrees tilt on the antenna.

Control and operation of the radar using the ERP and display menus on the CDU is covered under radar operation in this chapter.

MFD Displays

The display mode of the Collins MFDs and the selection of navigation course and bearing pointer sources are controlled by a combination of ERP keys and selectable menus on the CDU display pages. Additionally, the progress data block display on the MFD is enabled/disabled through the CDU. Those functions are discussed below.

Use of TCAS (TFC key) is not fully described in this manual. See mode control pages under radio tuning unit (RTU) in this chapter for a brief description of TCAS controls. Figure 16-23 shows an example of the TCAS traffic map display.

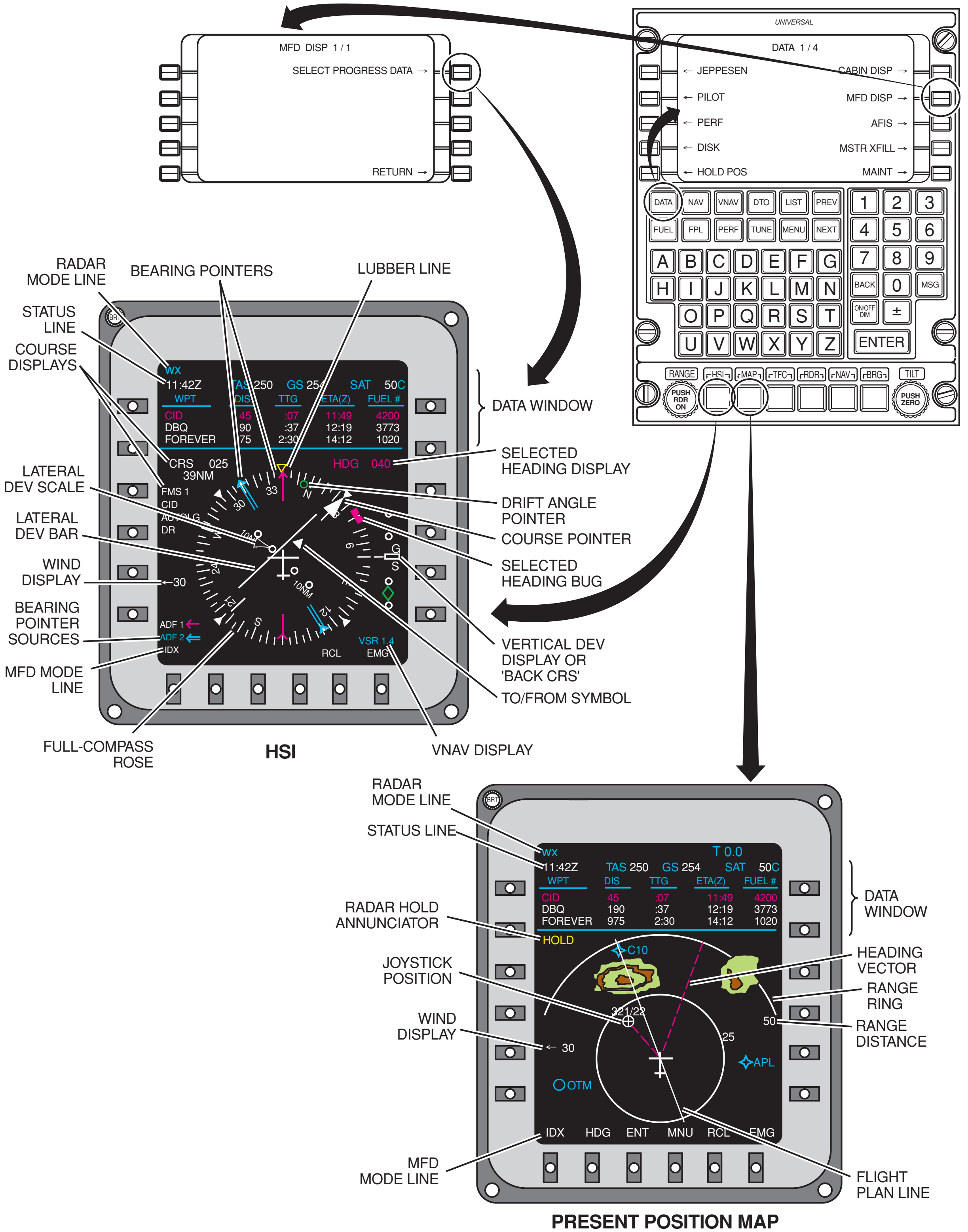


Figure 16-23. UNS-FMS CDU/ERP and MFD Displays (HSI, MAP, and TFC) (Sheet 1 of 2)

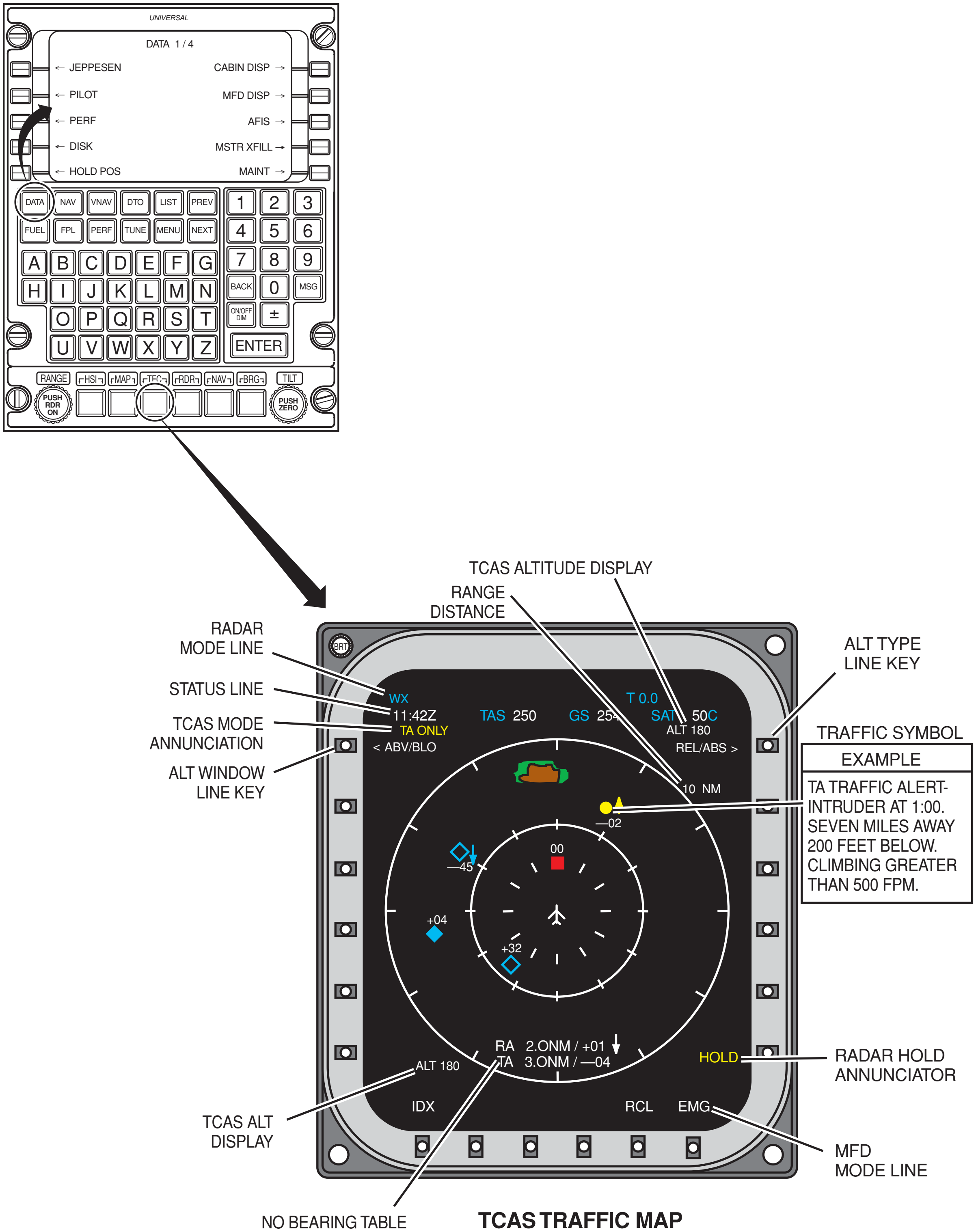
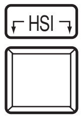


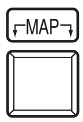
Figure 16-23. UNS-FMS CDU/ERP and MFD Displays (HSI, MAP, and TFC) (Sheet 2 of 2)

CONTROL KEYS (MFD DISPLAYS)



HSI

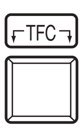
Press this ERP key to select the full compass rose display on the on-side MFD (see Figure 16-23). The HSI format simulates a conventional horizontal situation indicator. Weather radar can not be displayed in this format. See MFD in this chapter for more information on the MFD displays.



Map

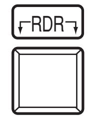
Press this ERP key to select the present position map display on the on-side MFD (see Figure 16-23). This display is a “heading-up” pictorial view of the relationship between the aircraft’s present position and selected navaids and waypoint data from the FMS database. Weather radar or terrain display (if installed) data can be superimposed on the present position map display as desired by the pilot.

See MFD in this chapter for more information on the map display.



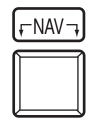
TFC

Press this ERP key to select the TCAS traffic map on the on-side MFD (see Figure 16-23). This map is a dynamic, “heading up” pictorial that shows nearby transponder- equipped aircraft. This screen displays traffic symbols that alert the crew to potential and predicted collision threats. The TCAS is optional equipment, and if not installed the TFC key will not be active.



RDR

Press this ERP key to display (on CDU screen) selectable radar modes and some available radar control functions (Figure 16-24). Depressing the RDR key a second time will display a second menu page. Radar controls are further described under Radar Operation in this chapter.



NAV

Press this ERP key to display the NAV SOURCE page on the CDU screen (Figure 16-24). This page shows the NAV sources that may be selected to supply active course information to the on-side PFD/MFD. Only one NAV source can be active on each side of the cockpit, although the pilot and copilot can have different active NAV sources selected. The selected NAV source appears in small green font, while the unselected sources appear in a large amber font with a prompt arrow. Press an adjacent line key to select a NAV source. The active NAV source is annunciated on the PFD and MFD to the left of the compass rose.

Possible NAV sources are VOR 1 (LOC 1 if ILS or LOC tuned), VOR 2 (or LOC 2), FMS 1, FMS 2, or MLS (if installed). If a single MLS is installed, only the MLS 1 prompt will appear. If dual MLSs are installed, then MLS 1 and MLS 2 will appear. Neither option will appear if MLS is not installed.

The NAV source for the SDU is selected on the SDU, not on the NAV SOURCE page.

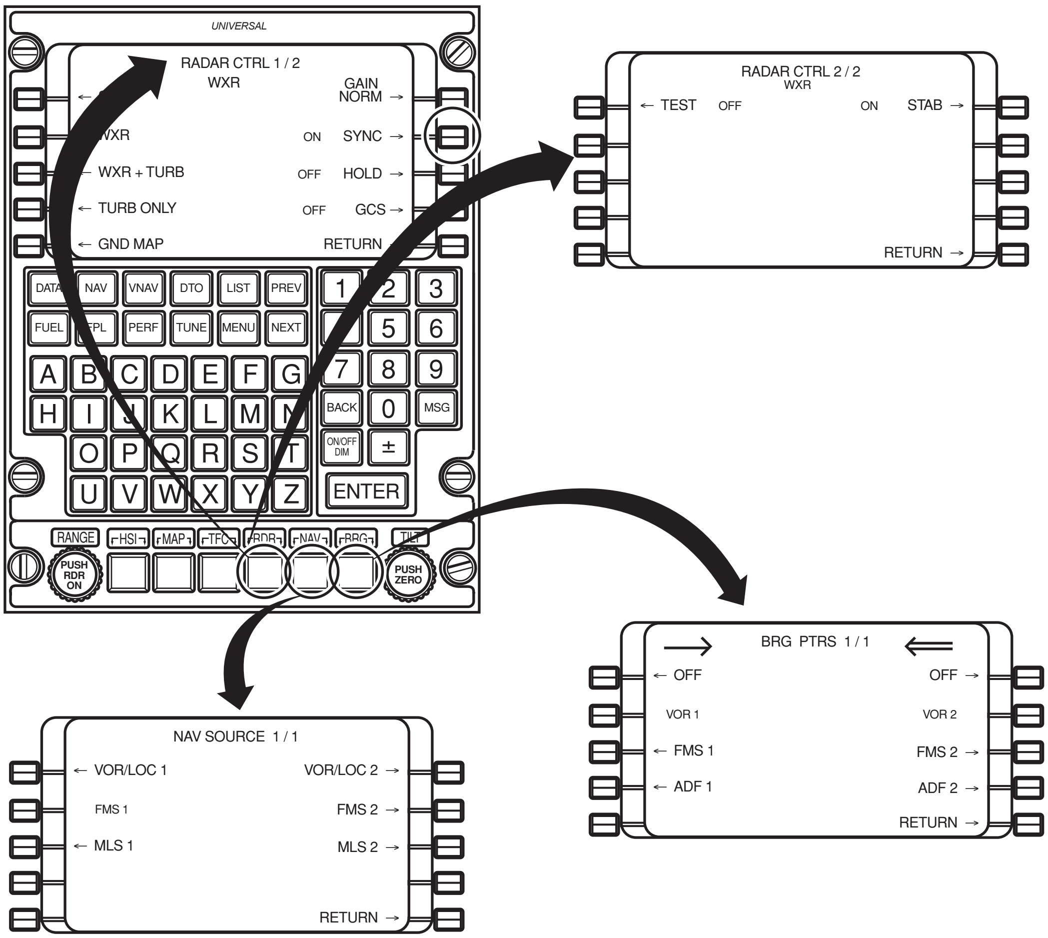
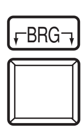


Figure 16-24. UNS FMS CDU/ERP, RDR, NAV and BRG Displays



BRG

Press this ERP key to select the BRG PTRS menu page on the CDU screen (Figure 16-24). This page shows the sources that may be selected to drive bearing pointers on the PFD and MFD. The left column lists sources for the single-bar pointer; the right column lists sources for the dual-bar pointer. The active source displays in green and is underlined; other sources are white. Press the adjacent CDU line key to select a pointer source. Depressing the PTR OFF line key removes the selected bearing pointer.

When a single source (ADF) is installed, this source is available to drive either bearing pointer. When dual sources are installed, a one or two suffix identifies each selectable source in the proper column. The bearing pointer source for the SDU is selected on the SDU and not on the BRG SRC page.

CDU ELECTRICAL POWER SOURCE

The left CDU receives DC electrical power through the ERP-AAP 1 circuit breaker, on the left CB panel, and the right CDU receives DC electrical power through the ERP-AAP 2 circuit breaker, on the right CB panel. Neither of the CDUs receive electrical power in the emergency bus mode of operation.

MULTIFUNCTION DISPLAY (MFD)

GENERAL

The LR 60 aircraft is configured with dual FMSs and two MFDs. The MFD has function keys on the bezel, around the tube, and a brightness control knob at the upper left corner.

The MFD is essentially an electronic replacement for the conventional HSI, but can also display a present position map, weather radar, TCAS, HSI, or TERRAIN display.

Basically, the same navigation information can be displayed on the MFD as can be displayed on the PFDs. Whatever NAV source or bearing pointer information is selected for display on the PFD will be the same on the on-side MFD with an appropriate display selected. Navigation signals come from the VOR, ILS, ADF, and FMS. The VOR, ADF, or FMS may be selected as the bearing pointer source using the BRG control key on the ERP to call up the BRG PTRS menu on the CDU. The active NAV is the navaid on which the course guidance and distance readout is based. VOR/ILS or FMS may be selected as the active NAV source using the NAV control key on the ERP.

Examples of the above are shown in Figure 16-24 (UNS CDU/ERP).

All NAV and display systems are monitored by the flight management computer. Invalid inputs are removed from the display, and when appropriate, warning flags are displayed.

The course heading panel (CHP) has two course set knobs that are used to individually select the courses on the pilot and copilot displays, and a heading set knob to set a single heading reference. If course guidance information is being displayed on the MFD, it will be the same as that selected for display on the on-side PFD. Also, the heading selected with the heading set knob will position the heading bug to the same heading on both PFDs and both MFDs.

MFD DISPLAY FORMATS

The HSI, MAP, TCAS, radar, and data window displays can all be displayed on the MFD. These functions are controlled through the on-side EFIS radar panel (ERP) and CDU.

The HSI, MAP and RADAR formats all have a radar mode line, status line, and data window at the top of the displays. See Figure 16-25 (UNS CDU). The TCAS format has a radar mode line and status line, but does not have a data window capability.

Radar Mode Line

The radar mode line indicates the radar mode of operation. Parameters that may be displayed on this line include: mode, ground clutter suppression (GCS), gain (G value), stabilization deselected (USTB), and tilt (T value).

Status Line

The status line displays the current time (Z), true airspeed, ground speed, and temperature (source and degrees). The FMS supplies the zulu time and GS data. The selected (normally on-side) ADC supplies TAS and SAT/RAT/ISA temperature data. Temperature format is normally SAT, but can be selected to RAT or ISA with the on-side ARP. This is the same as selecting the temperature format on the PFD. A readout is replaced by dashes or blanked if valid data is not available to these items.

DATA WINDOW

The data window on the MFD displays progress parameters for the next three waypoints in the flight plan. The data window is selected/deselected for display on the MFD DISP page. Depress the CDU DATA key to display the DATA 1/4 page. Then depress the line key adjacent to MFD DISP to display the MFD DISP page. On the MFD DISP page depress the line key adjacent to DESEL PROGRESS DATA or SELECT PROGRESS DATA.

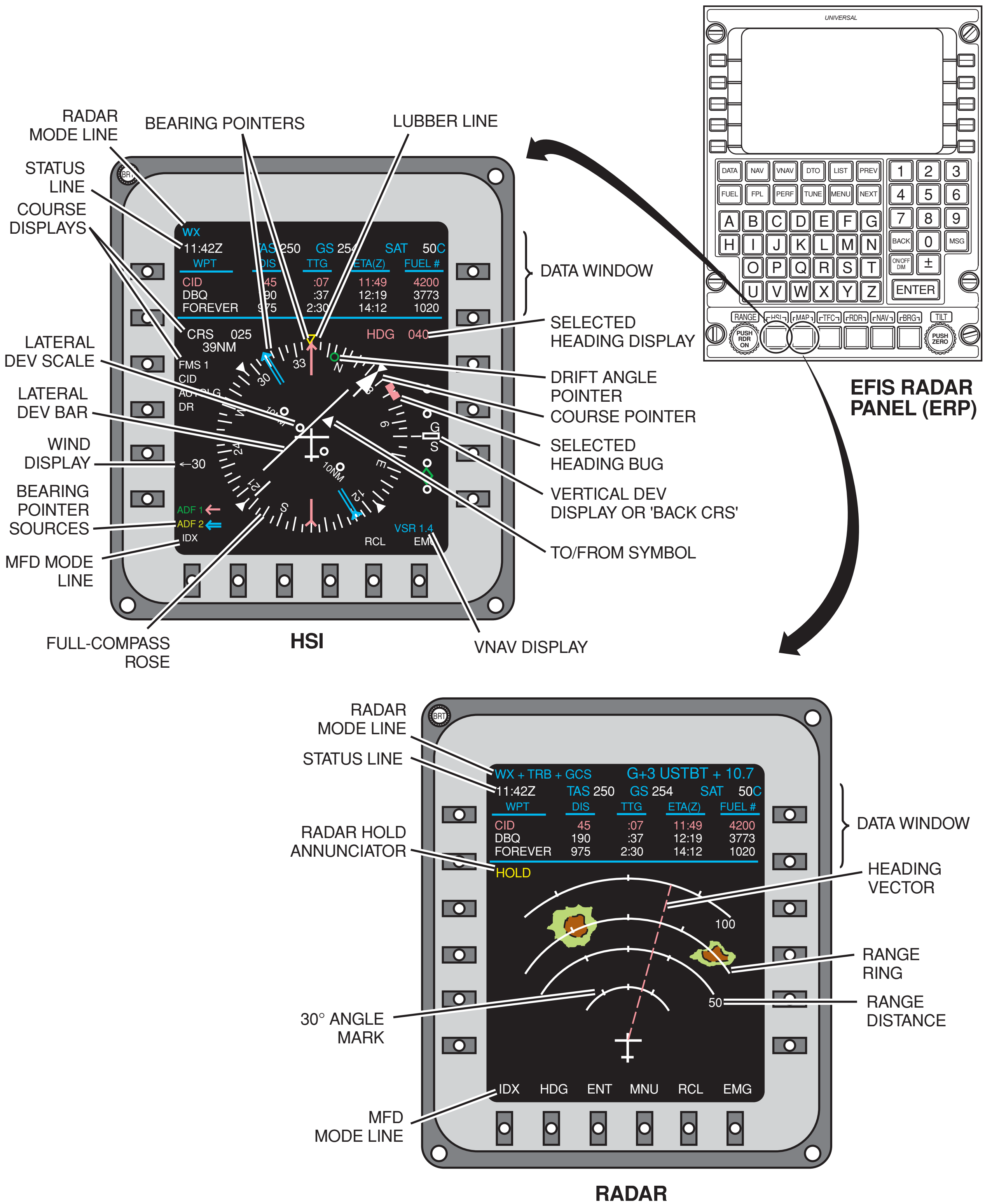


Figure 16-25. MFD with UNS CDU/ERP (Sheet 1 of 3)

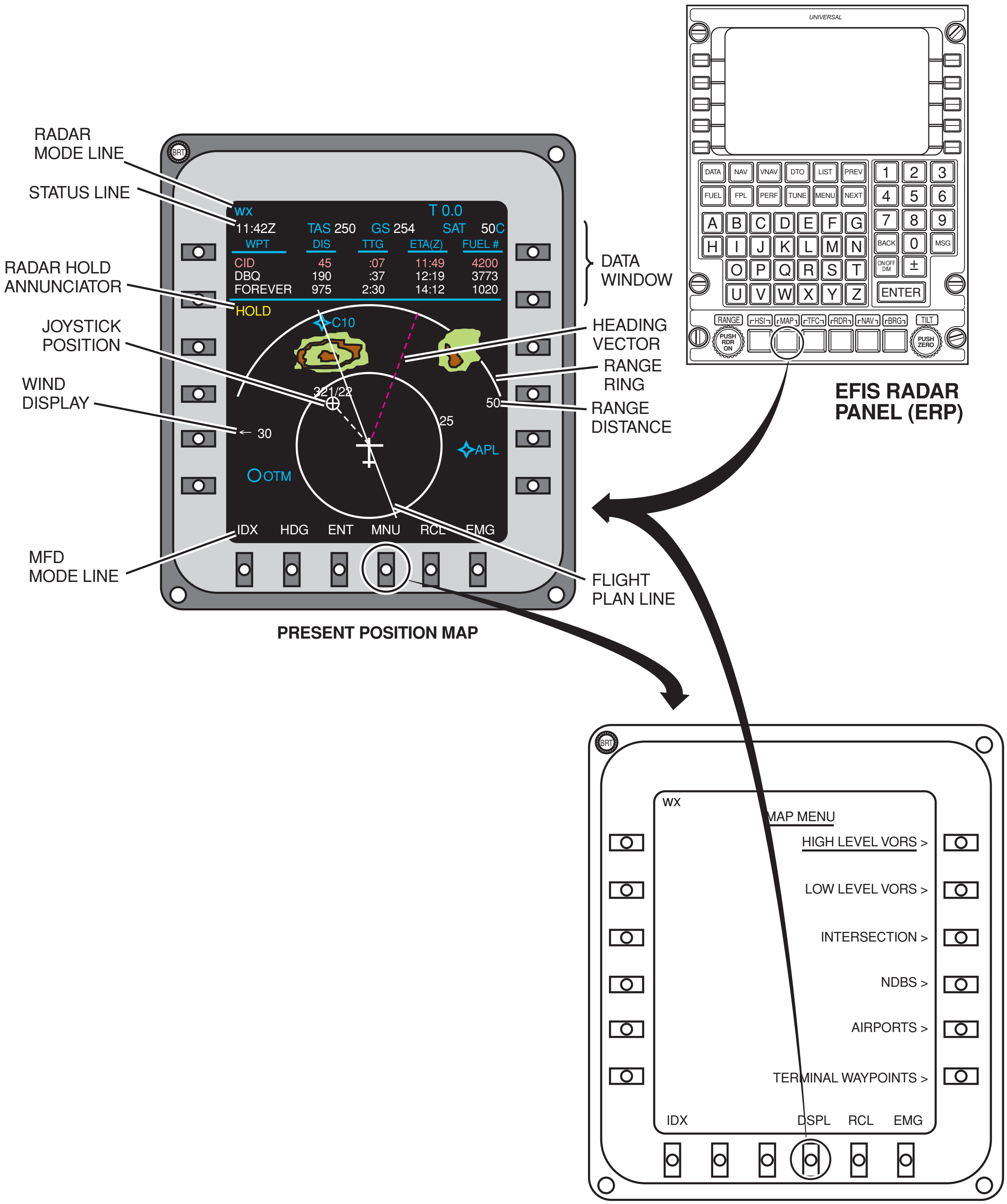


Figure 16-25. MFD with UNS CDU/ERP (Sheet 2 of 3)

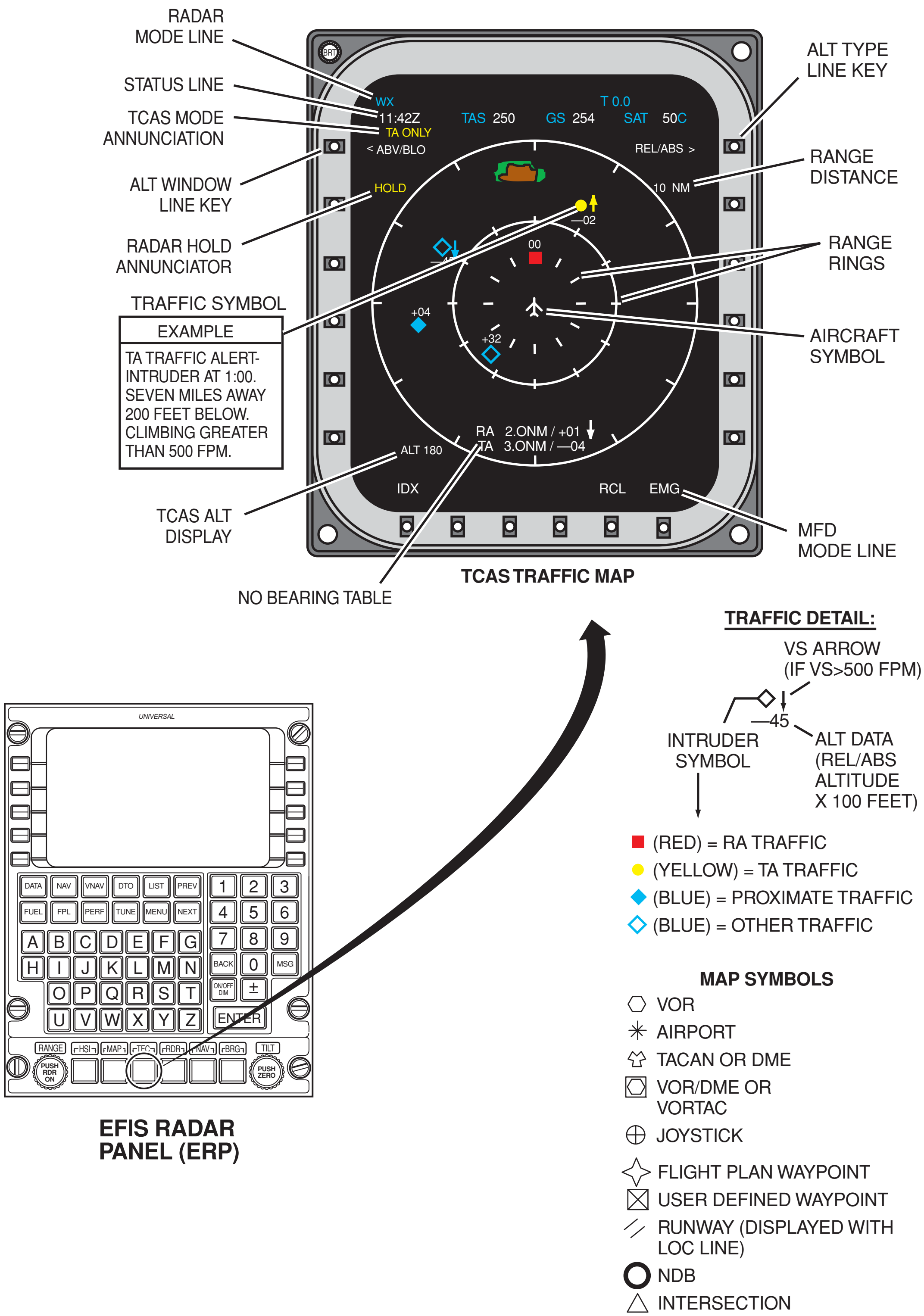


Figure 16-25. MFD with UNS CDU/ERP (Sheet 3 of 3)

The progress parameters include name (WPT), distance to the waypoint (DIS), time to go (TTG), estimated time of arrival (ETA), and pounds of fuel remaining at the waypoint (FUEL #).

HSI FORMAT

The HSI display is selected with the HSI control key on the CDU/ERP as described under Universal UNS CDU/ERP Operation/ Description (see Figure 16-25). The HSI display on the MFD, is very much like and repetitive of the HSI display on the PFD, except it is a full 360° compass rose. Active NAV source, bearing pointer select, course set, and heading set are all set the same on the on-side MFD as they are on the PFD. Under normal conditions, most pilots will not use the HSI display very much since most of the information on it is also displayed on the PFD and SDU. The MFD can better be utilized for other functions such as radar, map, TCAS, checklist (MFD only) or TERRAIN display.

In addition to the fact that it is a full compass rose, the MFD HSI display has some other minor differences from the HSI display on the PFD.

Wind Vector and Speed Readout

The HSI display on the MFD has a wind vector arrow and speed readout to the left of the compass rose. This wind display will only be present if the on-side FMS detects a wind in excess of five knots.

Vertical Deviation Display

There is a vertical deviation display on the right side of the compass rose that is basically a repeat of the vertical DEV display on the right side of the PFD attitude display and works under the same conditions.

VNAV Display On HSI Format

A VNAV display may be added to the HSI format. The VNAV display is selected from the VNAV page on the CDU. VSR X.X appears in the bottom right corner of the screen when VNAV display is selected and no vertical path is defined (on CDU). This value (in thousands of feet per minute) is the current vertical speed required to fly directly to the vertical waypoint.

VNAV operation is covered in the applicable FMS *Pilot's Guide*.

PRESENT POSITION FORMAT

The present position map display is selected for display on the MFD by depressing the MAP control key, on the on-side ERP (see Figure 16-25).

The map display is a dynamic geographic pictorial of the flight as it occurs. The map display is always centered on the aircraft's present position, with current heading toward the top of the screen. The screen shows a moving-map display relative to the aircraft's present position.

The MAP display has a wind vector and speed readout on the left side of the screen, the same as on the HSI.

On the MAP display (including radar and VNAV overlay), two range rings provide a distance measuring scale. These rings show distance from the aircraft. A numerical distance label displays by each range ring. Rotate the RANGE knob on the CDU/ERP to change the display range. A range of up to 300 NM is available on the MAP displays.

On the MFD, the next 15 waypoints in the flight plan will be displayed (if in range) plus a variety of background navaid symbols may be selected or deselected for display on the map. With the present position map or the plan map displayed on the MFD, depressing the MNU line key at the bottom of the MFD calls up the MAP MENU display (Figure 16-25), which is a menu of items (high altitude VORS, low altitude VORS, etc.) that can be selected/deselected for display on the MFD MAP display. These items are selected/deselected with the adjacent line keys. The item labels are underlined and green when selected and white when deselected.

The flight plan displays on the map as a solid white line, even if FMS is not the active NAV source. This track line consists of straight line segments connecting consecutive flight plan waypoints. The entire flight plan (up to 15 waypoints) can be displayed on the MFD if within the range selected on the display. The TO waypoint displays in magenta.

MAP and RADAR Display

Radar targets may be superimposed over the map display. To add radar to the map display, press the PUSH RDR ON switch in the left control knob on the CDU/ERP; to remove radar display, press the PUSH RDR ON switch a second time. To display the radar only display (see Figure 16-25), first select the HSI display on the MFD and then depress the PUSH RDR ON switch.

RADAR FORMAT

This display shows the weather radar situation without any distracting background data. This display presents a forward view, ±60° of the aircraft heading. There are four range rings to provide a distance measuring scale. Numeric distance labels display by two of the range rings. The range is changed with the RANGE knob on the CDU/ERP and is selectable to 300 NM.

Also, on the RADAR only display, three angle marks display on each range ring. These marks provide a directional scale relative to aircraft heading. The angle marks to the left and right of center show 30° left and right of the aircraft heading.

TCAS FORMAT

Press the TFC control key on the CDU to display the TCAS traffic map. This map is a dynamic heading-up pictorial that shows nearby transponder equipped aircraft. This screen displays traffic symbols that alert the crew to potential and predicted collision threats.

Note that navigation, background navaid symbols, and joystick functions are not available for display on this map.

Radar targets may be superimposed onto the traffic map display. To superimpose radar on the traffic map, press the PUSH RDR ON center switch in the range knob (on ERP). If radar is displayed on the traffic map, press the TFC control key (again) to remove the radar overlay or press the PUSH RDR ON switch to replace the traffic map with the Radar Only display.

MFD ELECTRICAL POWER SOURCE

The MFDs receive DC electrical power through the MFD 1 circuit breaker, on the pilot CB panel, and the MFD 2 circuit breaker on the copilot CB panel. The MFDs do not receive electrical power in the emergency bus mode of operation.

MULTIFUNCTION DISPLAY (MFD) (ADDITIONAL FUNCTIONS)

GENERAL

The MFD is capable of performing additional functions and providing much more data including an aircrew checklist. These functions and data displays are controlled through use of the bezel keys on the MFD. There are six keys on each side of the MFD and six across the bottom.

Joystick

A joystick capability exists with the MFD(s). If two FMSs and two MFDs are installed in an aircraft there is still only one joystick, but it can be used on either MFD. With dual MFDs there will be a selector button in front of the

joystick to connect the joystick to the left or right MFD (Figure 16-26). The joystick is located on the left end of the CHP, which is on the center console just forward of the CDUs. The joystick can be used to enter a waypoint or position into the FMS, used to change pages/ chapters on the checklist display, or to change pages in the MFD route library.

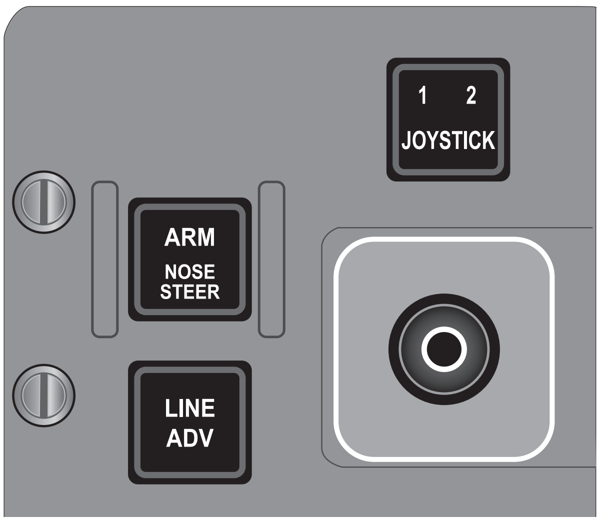


Figure 16-26. Joy Stick/Line Advance Controls

When the present position map or plan map is being displayed on the MFD, the joystick symbol (⊕) will appear on the display if the joystick select button is selected to the corresponding side and the joystick is moved. The symbol will move on the map in response to the joystick movement. The azimuth and distance from the aircraft to the position of the joystick symbol will also appear on the display. If the FMS is set up to receive the information, the waypoint can be entered into the FMS as a waypoint by depressing the ENT key at the bottom of the MFD. After approximately 20 seconds of inactivity on the joystick, a timeout circuit will remove the joystick symbol from the display. Subsequent movement of the joystick will cause the symbol to reappear.

To enter a joystick position into the FMS, complete the following steps:

1. Move joystick to position on MFD.
2. An ENT message will appear at the bottom center of the MFD; depress ENT.
3. Depress DTO, LIST, and select R1.
4. If it is desired to place the point in the flight plan, select the point in the flight plan where the entry is desired. Depress LIST and an R1 point will appear; select this point.
5. Subsequent entries are indicated R1, R2, etc.

MFD CONTROLS AND DISPLAYS

IDX

The IDX key is the first key on the left of the bottom row of keys. Depressing this key accesses the FMS remote data pages through the INDEX PAGE (Figure 16-27). This INDEX PAGE is the top level MFD menu.

NOTE

In a dual FMS system, if cross-side FMS is selected as the active NAV source, then the index page cannot be displayed. All selections made from the index page are not available.

MFD Mode Line

This line labels the bottom row of MFD line keys according to available functions. The RCL and EMG functions are available on the index page. The IDX label also displays on this page for consistency.

Possible MFD bottom line key labels are IDX, HDG, ENT, up ▲ and down ▼ line scroll arrows, MNU, DSPL, SKP, RCL, and EMG. These line keys are briefly described below.

Press IDX to display the index page. Press HDG to toggle the heading vector display

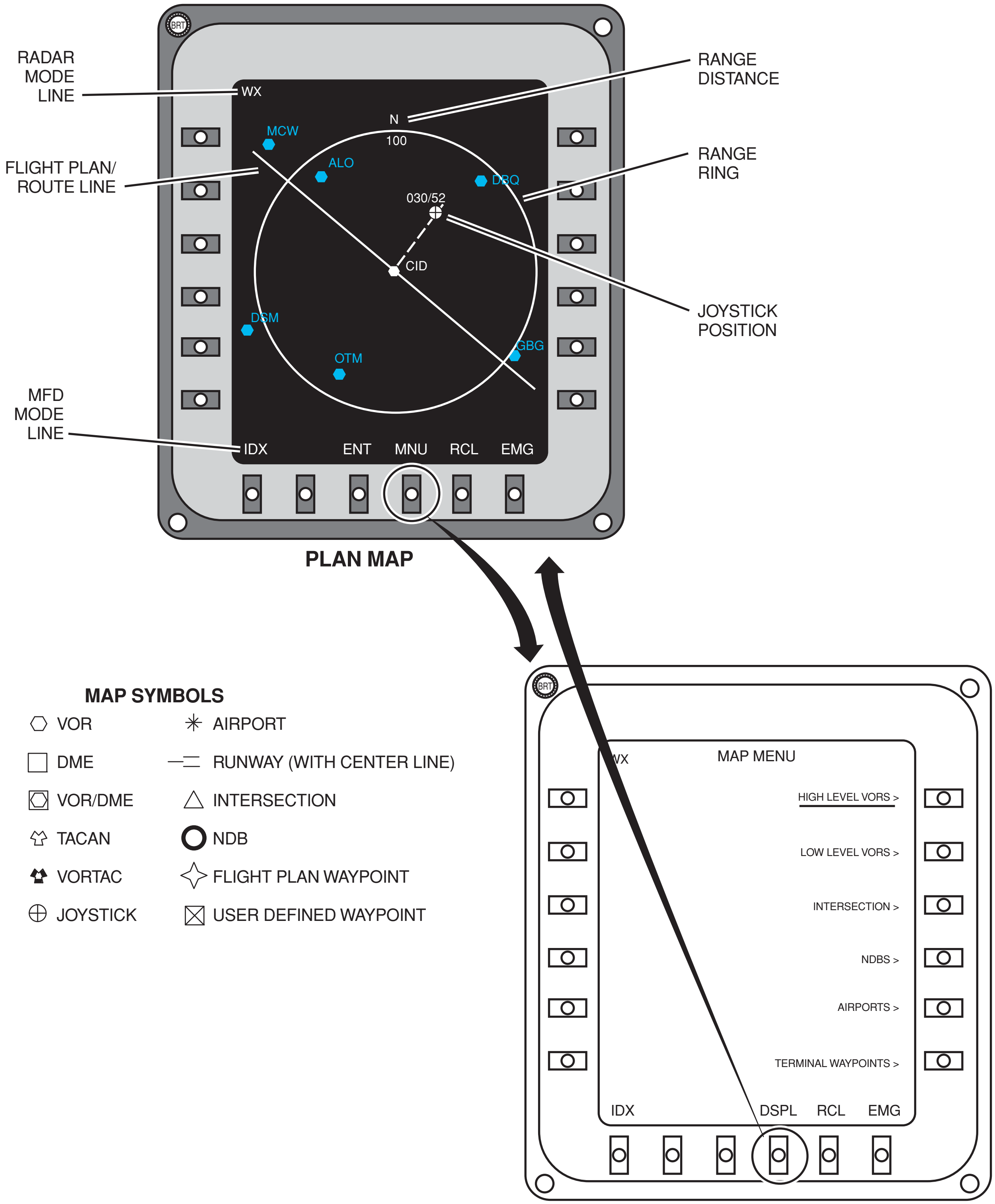


Figure 16-27. MFD with INDEX PAGE Menu and Displays (Sheet 1 of 2)

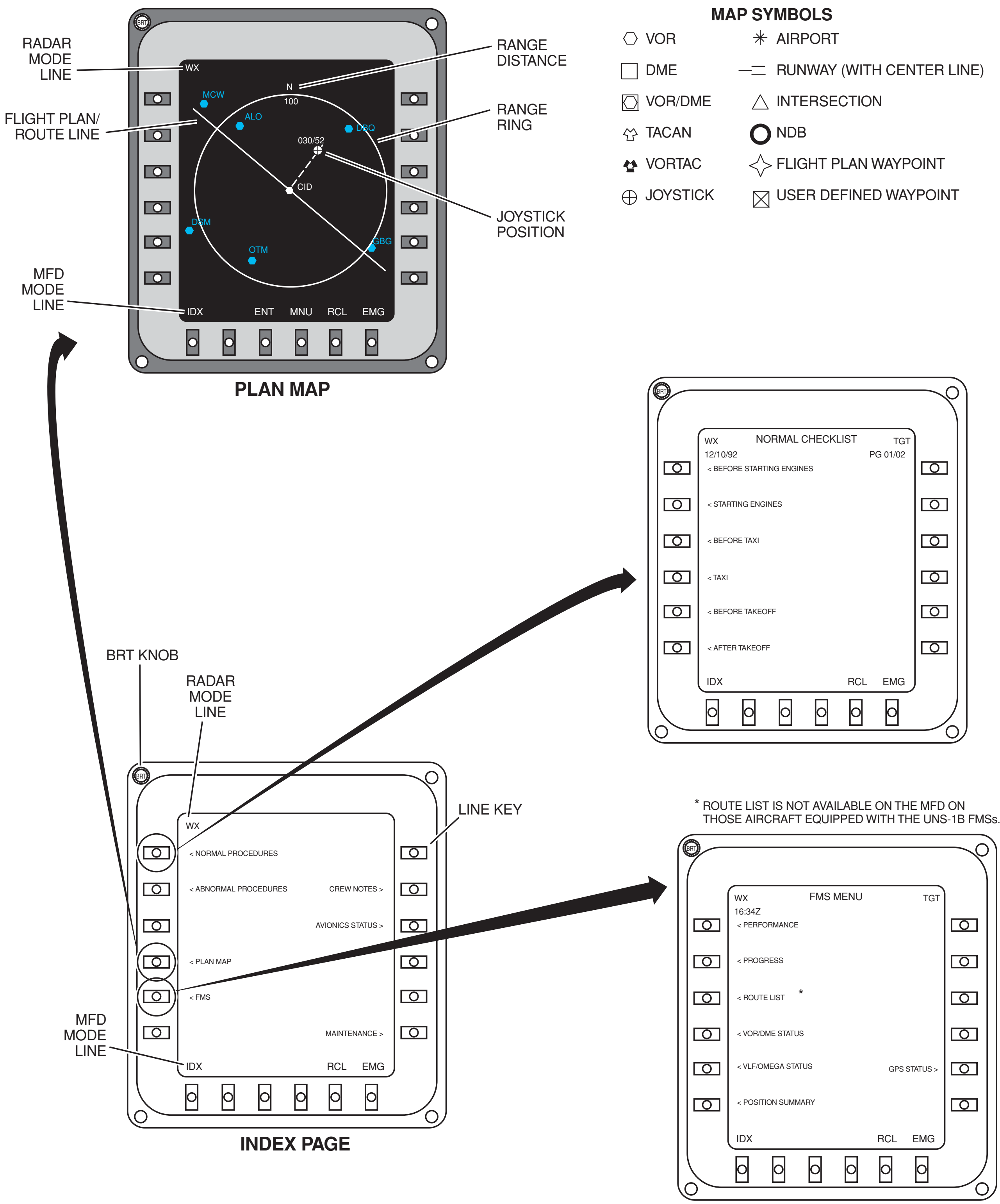


Figure 16-27. MFD with INDEX PAGE Menu and Displays (Sheet 2 of 2)

on/off. Press an up/down arrow key to scroll page data. Press MNU to display the previous menu (or the map menu). Press DSPL to display the present position or planning map from a map menu page. Press SKP to skip lines of page data (checklist). Press RCL to recall the previously used checklist (if not currently in a checklist format) or to find the first unchecked item in the present checklist. Press EMG to display the emergency checklist menu.

NORMAL PROCEDURES

Line Key

Press the NORMAL PROCEDURES line key on the MFD index page to display the normal procedures checklist page (Figure 16-27) if a checklist has been previously installed. The checklist must be installed by the operator; it does not come installed with delivery of a new aircraft.

The manufacturer of the FMS can provide computer programs that may be used to create a checklist on an IBM-compatible PC with a 3.5 disk drive. The checklist is then entered into the FMS using the DTU.

The first time each flight (after each power up) that a normal, abnormal, or emergency checklist is called up, the pilot must acknowledge the preamble by pressing the upper-right line key on the MFD.

After acknowledging the preamble (if necessary), a menu shows the available normal procedures checklists. Press the adjacent line key to display the desired checklist.

A checklist displays in three colors. Unchecked lines are blue, the cursor line is magenta, and checked lines are green. Active controls are the bottom row of MFD line keys, the CHP joystick, and a remote line advance switch on the center pedestal.

MFD mode line (bottom row of keys) labels on the NORMAL CHECKLIST page are as shown in Figure 16-27. If a specific checklist is selected from this menu, the mode line will also include the line up ▲ and line down ▼ symbols and SKP (line skip).

The CHP joystick may be used with checklist pages. If dual MFDs are installed, set the joystick select switch, located in front of the joystick, to desired (MFD 1 or MFD 2) position. Move the joystick aft to display the next checklist page. Move the joystick forward to display the previous checklist page. Move the joystick right to exit the checklist, and select the next checklist on the normal procedures menu. Move the joystick left to exit the checklist, and select the previous checklist on the normal procedure menu.

A remote line advance button is mounted on the center pedestal to the left of the joystick (see Figure 16-26). Press the line advance button to check the cursor line, and scroll down one line.

If it is necessary to leave the checklist page on the MFD, and then you wish to return to it, depressing the RCL key on the MFD mode line will return you to the last page of the checklist that was in use.

ABNORMAL PROCEDURES

Line Key

Press this line key to display the abnormal procedures checklist page if checklist was previously installed. A menu shows the available abnormal procedures checklists. The format and operation is the same as described above for the normal procedures checklist.

PLAN MAP Line Key

Press this line key on the MFD index page to display the plan map (Figure 16-27). The plan map is a static true-north-up map, that geographically shows a section of the flight plan or a selected route. The plan display provides a convenient way to plan and map a deviation from the flight plan or any route.

The center of the map depends on what is currently being displayed on the on-side CDU. When a flight plan page is displayed, flight plan waypoints are shown and the map is centered at the middle waypoint on the CDU page. When a route page is displayed, route waypoints are shown and the map is centered at the middle waypoint on the CDU page.

NOTE

To scroll in plan map view, display the FPL page and highlight a waypoint. The center of the plan map will be just prior to the point selected.

The plan map shows the selected navaids within range and the flight plan/route waypoints.

A variety of navaids may be selected or deselected for display on the plan map. The navaids to be displayed are selected from a map menu page. Press the MNU line key to access this page. Figure 16-27 depicts available navaid selections. On the map menu page, press the MFD line keys to select/deselect the desired navaids for display. After the navaids are selected, press the DSPL line key at the bottom of the MFD to display the plan map.

To avoid too much clutter and overlap on the plan map, it is recommended that the number of navaids selected be kept to a minimum and that the plan map range be kept as low as possible. The plan map range is controlled by the range knob on the CDU/ERP and has a capability up to 600 NM.

NOTE

Radar cannot be superimposed on the plan map.

FMS Line Key

Press this line key on the MFD index page to display the FMS MENU (see Figure 16-27). The FMS MENU is not operational at this time on those aircraft equipped with the UNS FMS.

CREW NOTES Line Key

Press this line key on the MFD index page to display the CREW NOTES page. This page displays a user defined checklist or text that has been entered into system memory.

AVIONICS STATUS Line Key

Press this line key on the FMS index page to display the AVIONICS STATUS page.

MAINTENANCE Line Key

Press this line key to display the MAINTENANCE page. A menu shows maintenance parameters available for display.

The FCS diagnostics function can be accessed using the MAINTENANCE line key. When selected, the FCS Diagnostics option will appear next to the same line key. When the adjacent line key is depressed, instructions will appear on the MFD telling how to access the FCS diagnostics page.

RADAR OPERATION

GENERAL

The WXR 840 (receiver/transmitter/antenna) is X-band radar that detects atmospheric moisture and ground feature returns in front of the aircraft. The TWR-850 is the same, but also detects air turbulence. Both of these models have low wattage (24 watts) transmitters combined with highly sensitive receivers. This feature reduces the hazardous area in front of the antenna to approximately 2 feet.

RADAR CONTROLS

The radar controls are through the CDUs and ERPs. The radar is controlled through two knobs on each ERP, the RDR key on the ERPs, and through menu selections on the CDU screens (Figure 16-28).

Either CDU/ERP can control the radar with Sync Off selected. The CDUs/ERPs can independently control the radar, commanding different modes, ranges, tilt, etc. at the same time.

The weather radar situation can be displayed on both MFDs. It can be superimposed over the MFD map display or it can be displayed alone without any distracting background data.

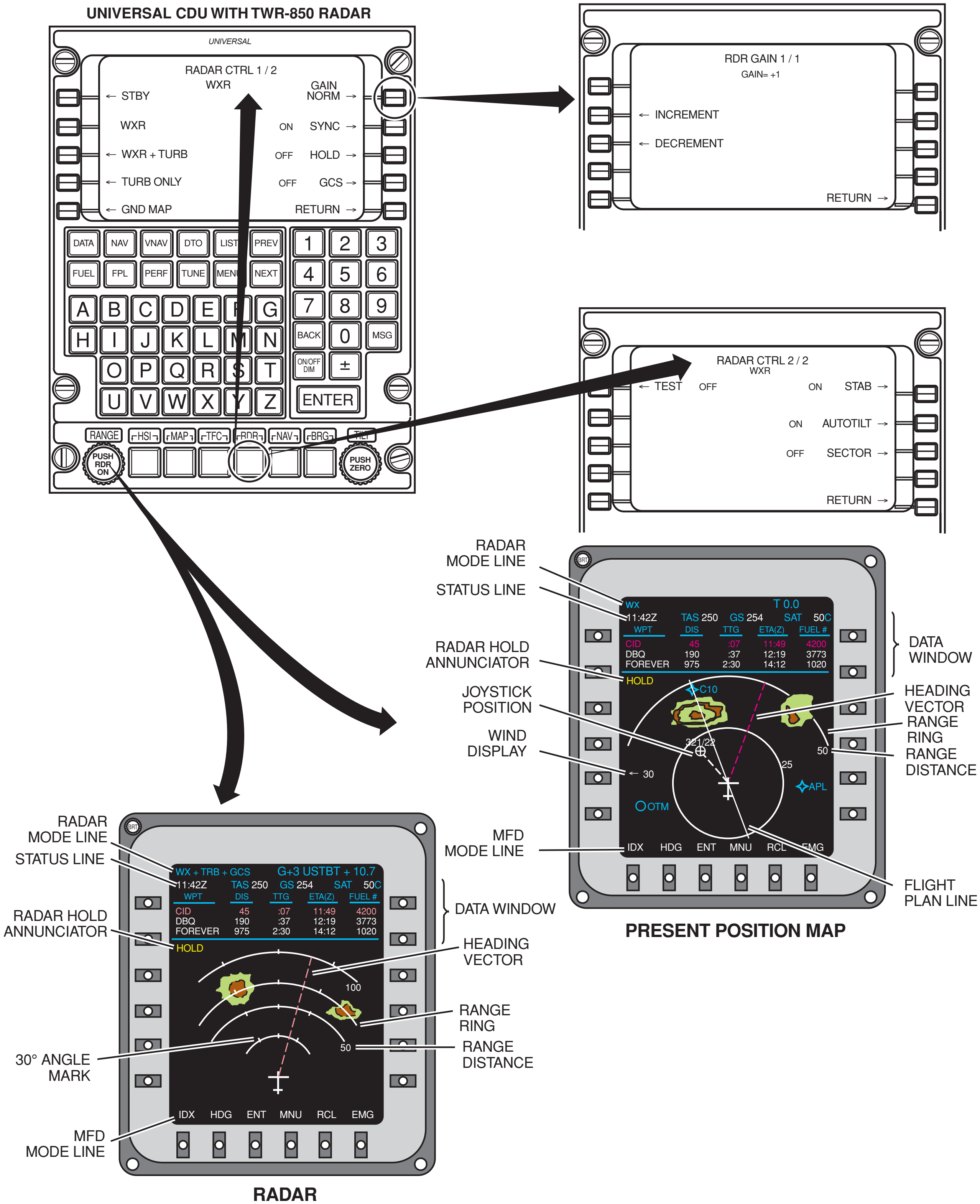


Figure 16-28. Radar Functions and Displays

RADAR MODE CONTROLS

FMS

Pressing the RDR key on the CDU causes the RADAR CTRL page (see Figure 16-28) to display on the CDU (WXR-840). On those aircraft equipped with the TWR-850, depressing the RDR key, on the CDU, causes the first of two RADAR CTRL pages to display.

The modes that may be selected by depressing the adjacent line key are: STBY, WXR, WXR+TURB, (TWR-850 only), TURB ONLY (TWR-850 only), GND MAP, or TEST. The active mode selected is in small green font, and other available selections are in a larger amber font. The selected mode is also annunciated at the top of the MFD display pages.

STBY

Selecting STBY turns the radar transmitter off while the other modes turn the transmitter on. STBY automatically selects after touchdown with the UNS FMS installation.

WXA

This is the radar's basic mode.

WXA+TURB

This mode is only available with the TWR-850 and it enables the radar to detect both weather and turbulence, but is limited to selected ranges of 50 NM or less.

Turb Only

This mode also is only available with the TWR-850. This mode displays turbulence that is related to precipitation; it does not display clear-air turbulence. The TURB ONLY mode can be selected for 30 second intervals only. If TURB ONLY is selected again before the 30 second timeout, WX+TURB is enabled. Any other mode can be selected before the 30 seconds has elapsed. At the end of the 30 second timeout period, the selected mode will automatically change to WX+TURB mode. TURB ONLY mode is also limited to a maximum of 50 NM range.

GND Map

This mode causes the radar to display prominent terrain and ground features instead of weather, and the color format changes in this mode. When GND MAP mode is selected, the GND CLTR SPRS option and its associated ON/OFF selection is not available and is removed from the RDR functions menu.

Test ON/OFF

When ON is selected, a test pattern showing all four colors is displayed. Test mode is automatically cancelled if the menu page is changed.

OTHER RDR CONTROL FUNCTIONS

Radar Display

The radar display (display ON–OFF) is controlled with the PUSH RDR ON switch in the left control knob on the CDU/ERP. Press the PUSH RDR ON switch once to display the radar on the MFD.

If the present position map is being displayed, depressing the PUSH RDR ON switch once will cause the radar picture to overlay the map. Depressing the PUSH RDR ON a second time will remove the radar picture.

To display the radar only display (see Figure 16-25), first select the HSI display on the MFD and then depress the PUSH RDR ON switch. When the radar only display is on the MFD, depressing the RDR ON switch will cause the MFD to go to the MAP display.

The MFD displays have a radar mode line at the top of the screen. This line annunciates the radar mode or RDR OFF on the far left, the gain (G value), antenna stabilization turned OFF (USTB), ground clutter suppression (GCS), and the antenna tilt (T value). The word HOLD will appear in the upper left corner of the radar display while HOLD is selected to ON. RDR FAULT annunciates at this point if the RTA detects an internal fault.

The map and radar displays are described under MFD in this chapter. Examples are shown in Figure 16-28.

Range

The left knob on the CDU/ERP is used to control the range on the on-side MFD (both sides if SYNC is selected) any time radar is being displayed. The range is adjustable up to 300 NM except in the WX+TURB and TURB ONLY modes where it is limited to 50 NM. The same knob controls range on the map display and TCAS display when radar is not being displayed.

Tilt

The right hand knob on the CDU is used to control the antenna tilt and is functional in all radar modes except STBY. The range for antenna tilt is ±14°. Antenna tilt can be set to zero by depressing the PUSH ZERO TILT switch on top of the TILT knob. The antenna tilt angle is annunciated on the radar mode line at the top of the MFD display (e.g., T + 10.7). Autotilt is available on some models. See Autotilt in this section for more information.

Gain

The radar gain value is displayed on the CDU RADAR CTRL page and on the radar mode line at the top of the MFD display. The gain is adjustable through a range of ±3 and will read NORM when at the mid-level of gain adjustment. To adjust gain, depress the CDU line key adjacent to GAIN. This will display the RDR GAIN 1/1 page on the CDU. Depress the line key adjacent to Increment to cause gain to increase +1 for each time the key is depressed, and depress the CDU line key adjacent to Decrement to cause gain to decrease –1 for each time the key is depressed.

SYNC, HOLD, GND CLTR SPRS (GCS), and STABILIZATION

These are all alternate action switches; pressing the associated line key toggles between ON and OFF.

SYNC ON/OFF

When SYNC (ON) is selected on either CDU, the radar functions synchronize to what is selected on the cross-side CDU. From that point forward, the radar control functions of both CDUs behave as if they are fully parallel. Changing a radar function on one CDU will cause the opposite side to also change and both radar displays will be the same.

HOLD ON/OFF

When selected to the ON position, the radar image is frozen on the display and HOLD annunciates on the MFDs. The held display is not updated. When HOLD is selected to ON, it will automatically turn to OFF after 30 seconds and the display will return to the previously selected mode.

GND CLTR SPRS (GCS) ON/OFF

When ground clutter suppression is on, ground clutter is reduced, but so is sensitivity to low levels of precipitation. GND CLTR SPRS automatically turns off after 30 seconds.

AUTOTILT OFF/TILT

Selecting TILT (if installed) with the adjacent line select key on the second radar menu page enables the autotilt function of the radar. Autotilt is designed to reduce the pilot workload by automatically adjusting the current displayed tilt setting following altitude or selected range changes. Manual tilt commands from the TILT knob remain operational when autotilt is enabled. Autotilt may be enabled continuously as the system always uses the current manual tilt setting as the starting point.

When autotilt is selected on (TILT), it is annunciated with an A following the tilt setting on the weather radar mode line of the MFD (e.g., T+2.5A).

Stabilization (STAB) ON/OFF

This selection allows the antenna stabilization to be toggled on and off.

Target Alert, OFF/TGT

Target alert is available on some aircraft that are equipped with dual FMSs. Depress the line select key adjacent to TARGET ALERT, OFF/TGT to toggle target alert OFF and ON (TGT). When selected, weather targets that are detected will be annunciated without the targets themselves appearing on the weather radar display.

The target alert feature is intended as a pilot aid to reduce the chance of an inadvertent penetration of a thunderstorm or, on those aircraft with turbulence detection, an area containing precipitation related turbulence. The radar performs this task by conducting an automatic search of the air mass ahead of the aircraft. The radar starts by scanning at a 50 NM range while looking for precipitation related turbulence. This is followed by a scan at the 200 NM range looking for precipitation (targets). In the TGT/TURB search mode, radar parameters (mode, tilt, and range) are not controlled by the pilot. This creates the first TGT/TURB requirement: Before a channel of the radar can search for TGT/TURB, that channel must not be displayed to the crew. The reason for this restriction is that it is potentially confusing to display radar data that is continuously changing range, tilt, etc. In addition, a basic purpose of the function is to provide a "lookout" while the crew is involved with non-radar display functions (i.e., checklist or navigation data on the MFD).

Once TGT/TURB alert has been enabled, the FMS places either radar channel in search mode whenever that channel is not displayed. If either channel (or both channels) is in search mode, a blue TGT annunciator is displayed in the upper right hand corner of both MFDs. Detection of a precipitation target is annunciated by flashing an amber TGT annunciator in the upper right hand corner of the MFD.

On those aircraft equipped with turbulence detection, a turbulence alert is similar except that the annunciation is an amber TRB rather than TGT. If TGT and TRB alerts are received at the same time, the annunciation will alternate between TGT and TRB as long as both alerts are valid.

SENSOR DISPLAY UNIT (SDU)

GENERAL

The sensor display unit, mounted in the center instrument panel, between the RTUs, provides an electronic RMI display that replaces a conventional electromechanical RMI. The SDU uses a 3 inch by 3 inch, high-resolution, monochromatic display screen to display compass information (aircraft heading) and navigation data.

The SDU incorporates selectable formats and is able to display data from either VOR/ILS receivers, either DME, one or two ADF receivers, one or two VLF/ Omega receivers, one or two MLS receivers, and one or two FMS systems. All of the navigation inputs are fed to the SDU through a sensor display driver (SDD) that has two channels (left and right). Number 1 navigation receiver information is fed to the SDU through the left channel of the SDD and number 2 receiver information is fed through the right channel (Figure 16-29).

The display of navigation information is normally set up on the PFD and MFDs; however, it can also be displayed on the SDU as a backup.

In the emergency bus mode of operation, the PFDs and MFDs will be inoperative and the SDU will be the only remaining instrument on which an instrument approach can be made. Some aircraft are also equipped with a standby attitude gyro that incorporates glideslope and localizer information.

The heading information to the SDU is only provided through AHS 1, regardless of the AHS selection on the EFIS control panels. If AHS 1 fails in flight, heading information to the SDU will be lost and HDG in a box with

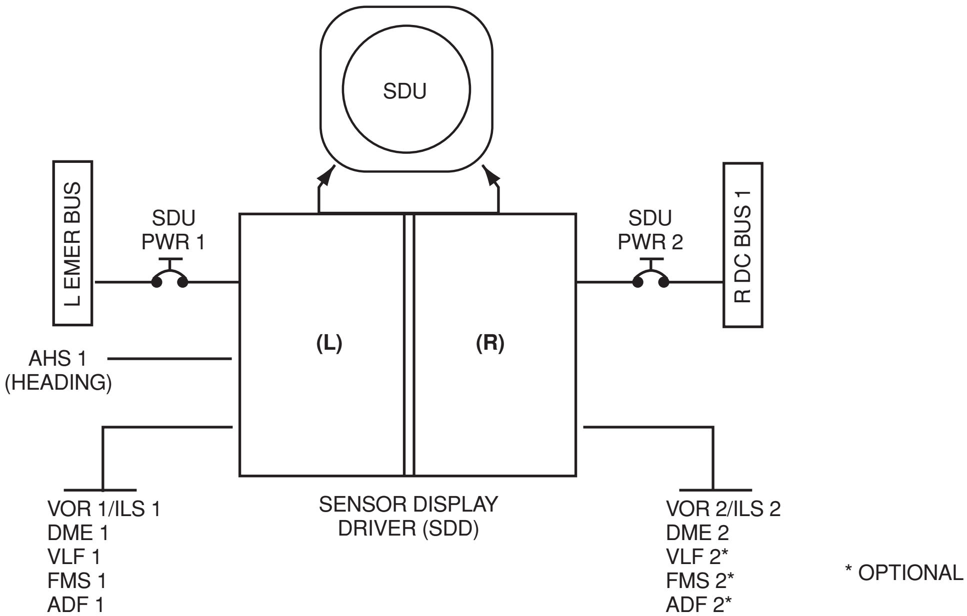


Figure 16-29. SDU Schematic

an X through it will be displayed at the top of the compass card.

The SDU has a brightness adjust knob in the upper left corner. The brightness is also controlled by the EFIS dimmer on the pilot INSTR LIGHTS panel. It is recommended that the brightness control not be operated any higher than necessary to avoid imprinting on the display screen.

There is a selector knob in the upper right corner of the SDU and one in each of the bottom corners for controlling SDU format and functions.

SDU FORMAT

The format knob in the upper right corner of the SDU is used to select the desired format on the SDU. The knob is rotated one direction to step through all of the possible selections, and when the last format option is reached, the knob has to be turned in the opposite direction to make further selections. An arrow will appear on the display when in the RMI format showing the direction the format knob should be turned (Figure 16-30, Sheet 1).

The current format selected is annunciated along the edge of the compass rose in the upper, left quadrant.

The available formats (not necessarily in this order) are RMI, VLF (VLF 1 and VLF 2 if two installed), VOR 1, VOR 2, DME, FMS 1 (also FMS 2 if two installed), and RMI again. LOC 1 or LOC 2 replaces VOR 1 or VOR 2 depending on the frequency that is selected in NAV 1 and NAV 2. In the RMI, VLF, and DME formats, VOR, ADF, or FMS bearing pointers may be selected for display with the two knobs on the bottom corners of the SDU. Figures 16-31 (Sheet 1 through 5) depict examples of the different formats. The FMS format is not shown, but is nearly the same as the VOR format. The only difference is that the FMS format does not have a course selection capability.

SDU ELECTRICAL POWER

Electrical power to the sensor display driver (SDD) is from two sources. The left channel of the SDD and the SDU receive DC electrical power from the L EMER BUS, through the SDU PWR 1 circuit breaker on the left CB

panel, and the right channel of the SDD receives electrical power from the R DC BUS 1, through the SDU PWR 2 circuit breaker, on the right CB panel.

If electrical power through the SDU PWR 1 circuit breaker (L EMER BUS) is lost, the SDU will be inoperative since there will be no power to the SDU display or to the left channel of the SDD. However, if power is lost through the SDU PWR 2 circuit-breaker, the left channel of the SDD will still function and the SDU will still function. Only receiver number 1 information will be available in this case. The SDU PWR 1 circuit-breaker receives power during the emergency bus mode of operation; therefore, left side (VOR/ILS 1, ADF 1, etc) navigation information will still be available (see Figure 16-20).

AUTOPILOT/FLIGHT DIRECTOR

GENERAL

The flight control system (FCS) includes the APS-850, which provides fail-passive autopilot (AP) and dual flight director (FD) functions. The FCS has two identical FCC-850 flight control computers (FCC) and three servos to move the elevator, ailerons, and rudder. The autopilot also uses the aircraft secondary pitch trim system to control the aircraft in the pitch axis.

A glareshield-mounted flight control panel contains the autopilot control panel in the center and identical FD mode select panels (MSPs) on each side (Figure 16-31). Either FD can be coupled to the autopilot.

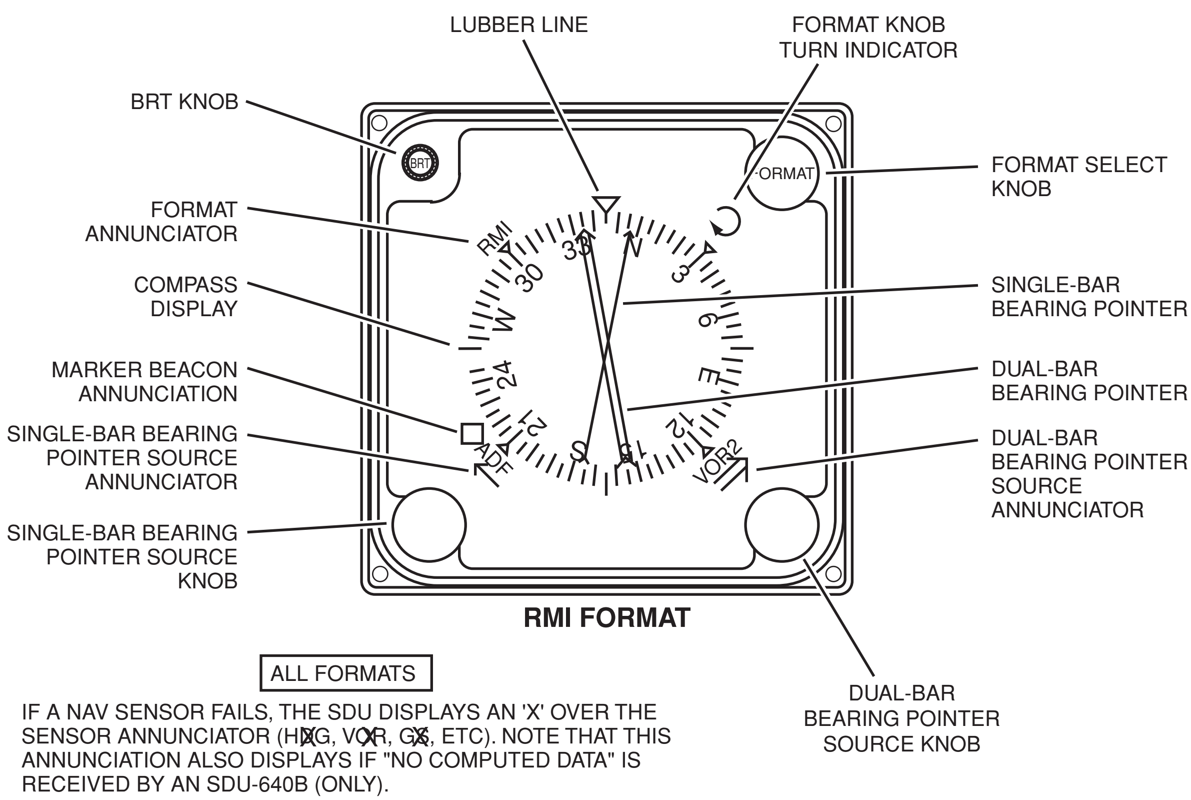


Figure 16-30. SDU Formats (Sheet 1 of 5)

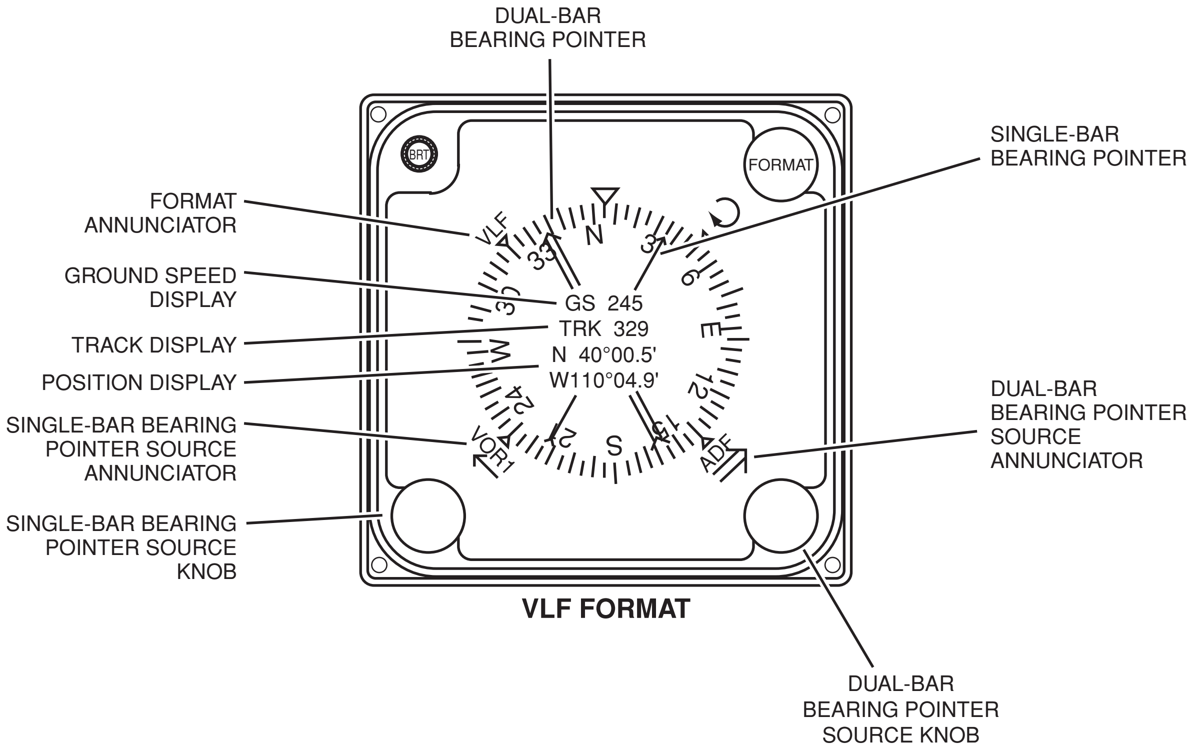


Figure 16-30. SDU Formats (Sheet 2 of 5)

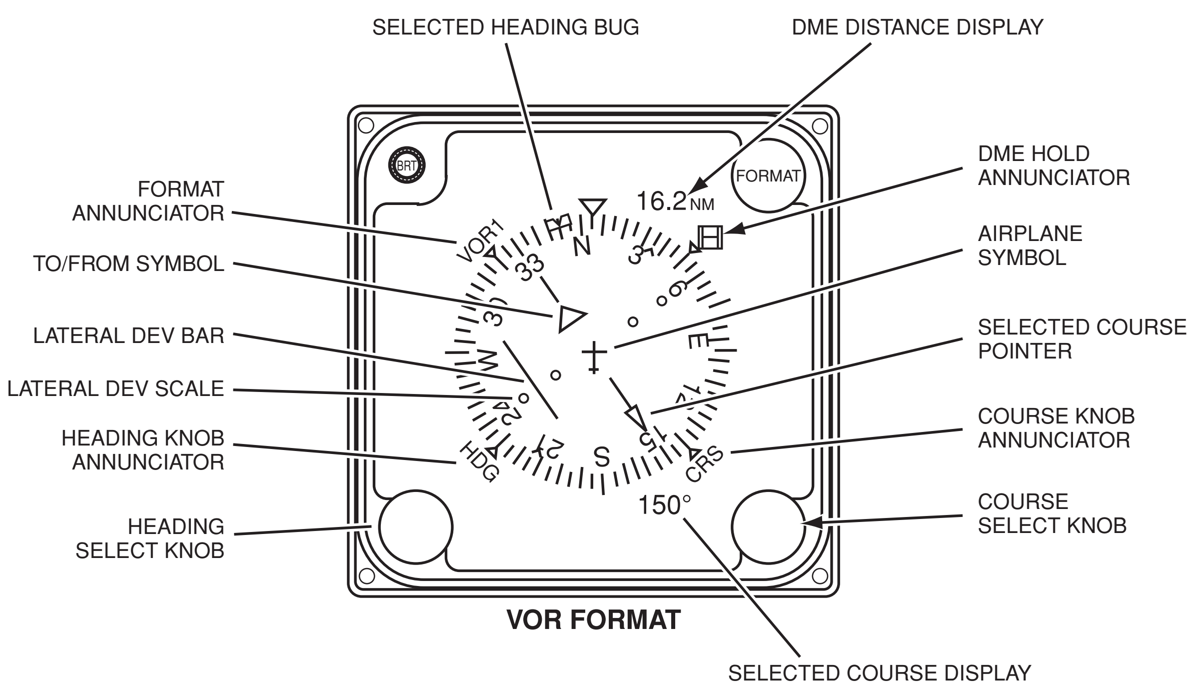


Figure 16-30. SDU Formats (Sheet 3 of 5)

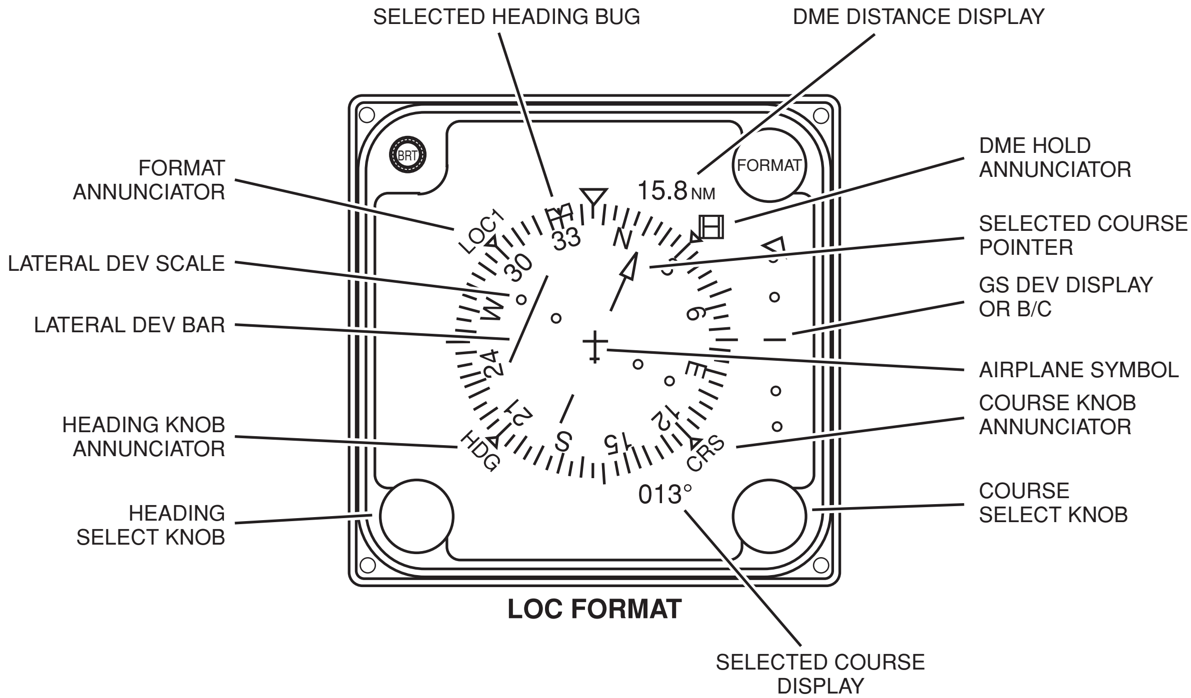


Figure 16-30. SDU Formats (Sheet 4 of 5)

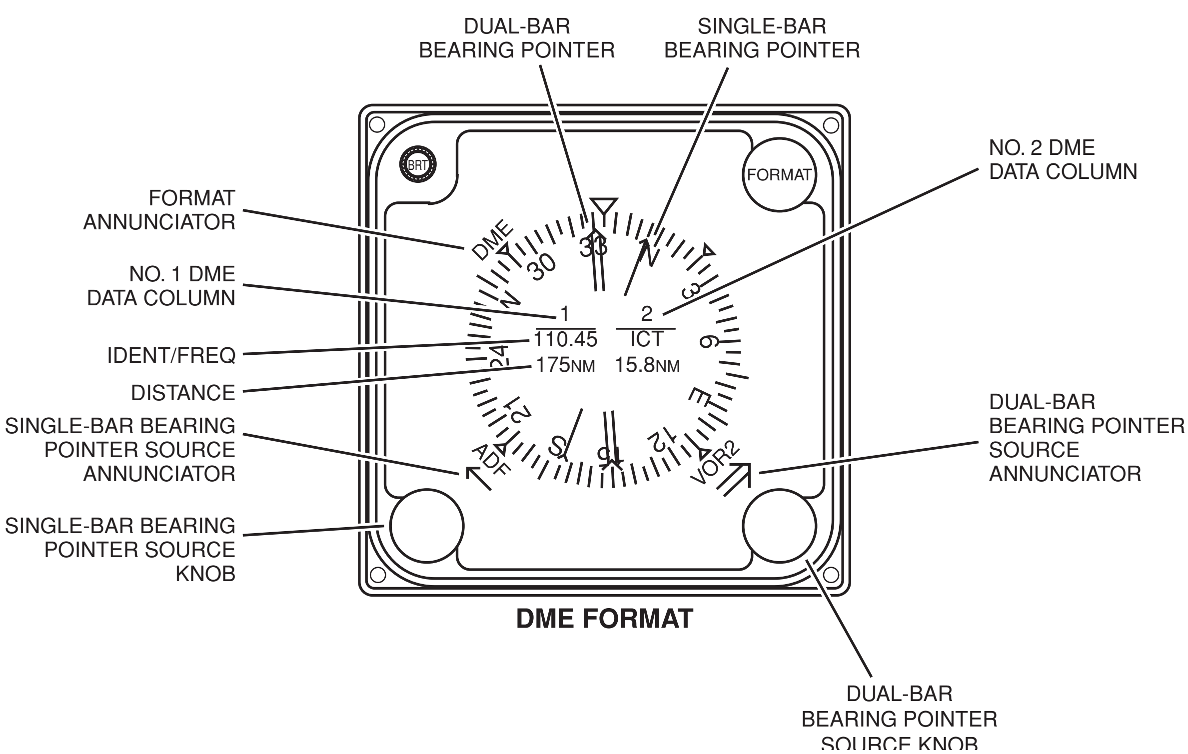


Figure 16-30. SDU Formats (Sheet 5 of 5)

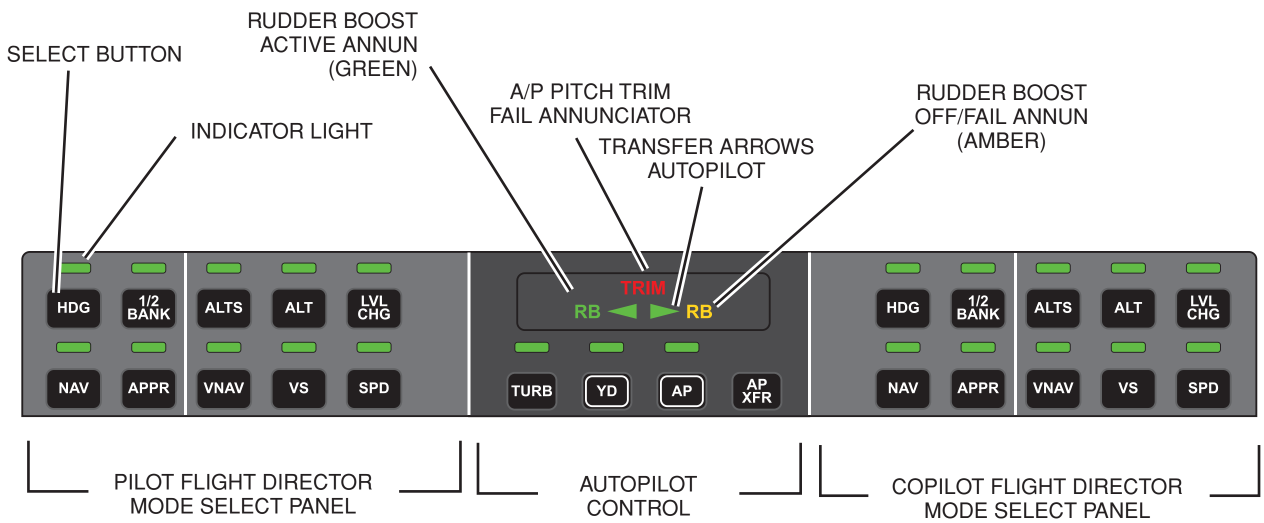


Figure 16-31. Flight Control Panel

The PFDs display flight director mode and autopilot information.

The two FCCs reside in the IAPS card cage located in the left nose, avionics bay. These computers provide independent flight guidance computation and operate together to provide 3-axis control (including yaw damper). The FCCs receive critical AHS data directly from the AHCs, and ADS air data through the IAPS concentrators. The autopilot and yaw damper will disengage and be inoperative if input from either AHS is lost, however, the FD on the side away from the failure will retain normal functions.

The system initiates a self-test sequence when electrical power is applied to the autopilot (avionics master switches both on). If the self-test sequence is not successfully completed, neither the autopilot nor yaw damper will engage, and a red FD flag will be displayed on the PFDs.

SERVO ACTUATORS

The pitch and roll servos position the elevator and ailerons in response to commands from the FCCs in the IAPS. When the yaw damper or rudder boost is engaged, the yaw servo positions the rudder in response to commands from the FCCs.

When the aircraft pitch is adjusted by the autopilot, the pitch servo is first energized to make an elevator input, and then the secondary pitch trim motor is activated to reposition the horizontal stabilizer until the effort of the pitch servo is canceled out. The autopilot trims the controls in the pitch axis, but does not trim the ailerons or rudder. The pilot should, therefore, insure that the aircraft controls are trimmed-up prior to engaging the autopilot. Also, wing fuel balance should be maintained to avoid an out-of-trim condition when the autopilot is engaged. See Chapter 15—“Flight Controls” for more information on the pitch and roll servos and secondary pitch trim.

AUTOPILOT (AP) CONTROLS AND OPERATIONS

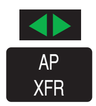
General

The flight control panel (FCP) contains a total of 24 selector buttons, all of which, except for AP XFR, have a rectangular green indicator light located above them to indicate when the associated select button is pushed ON. The AP control panel also contains a green RB (rudder boost) annunciator, a yellow RB annunciator, green autopilot transfer arrows, and a red autopilot pitch TRIM fail annunciator.

AP and FD modes are selected by depressing the applicable mode selector button on the flight control panel and FD mode select panel. The green indicator light above the select button illuminates to indicate a mode is selected. AP/FD mode engagement is also annunciated on the associated PFDs.

FD only mode selection is accomplished by depressing the applicable mode selector button without the AP engaged.

Engaged AP and FD modes may be disengaged by depressing the selector button a second time or selecting an incompatible mode. FD modes can also be deselected or cleared by depressing the F/D clear switch on the control yoke.

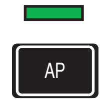


AP XFR

The AP XFR selector button is the only selector button that doesn't have a indicator light above it. Instead, the AP XFR function has two green arrows (one pointing right and one left) on the center of the AP control panel. One of these arrows will be illuminated to indicate which FD is coupled to the AP. There is also a green AP→ or AP ← (as appropriate) in the top left corner of each PFD attitude display to indicate which FD is coupled to the AP.

The autopilot defaults to the left side at power-up, but can be alternately selected to either side by depressing the AP XFR button. If AP XFR is selected to the right side when the autopilot is not engaged, the PFDs will display a white AP→ message.

AP Engage



AP

Press the AP button to engage the AP; an electrical interlock also engages the yaw damper. The indicator lights above the AP and YD buttons illuminate, the left or right transfer arrow illuminates, and the PFDs display green AP→ or AP ← messages.

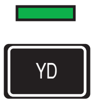
When the AP or YD buttons are depressed, engagement occurs only if aircraft attitude/rates are normal and if the FCCs do not detect any AP or YD failures. Autopilot pitch limits are +20° to –10°, and bank limit is normally 27°. Bank limit is 32° when using the roll command switch (1/2 bank automatically selects above 41,500 feet). AP roll rate and pitch rate limits are 5° and 3° per second respectively.

The pitch trim selector must be in the primary or secondary position for the AP to engage. Also, the AP will not engage when selected if secondary trim failure is detected by the FCCs.

When engaged, the AP flies the flight director commands from the coupled side. If no FD modes are selected on the coupled side and the AP is engaged, the AP will assume the attitude hold mode and maintain the attitude existing at the time of engagement.

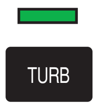
If roll attitude is within 5° of wings level when the AP is engaged and no FD lateral modes are engaged, the FCC generates commands to maintain the existing heading. If the AP is engaged with no lateral modes selected on the FD, and the roll attitude is greater than 5°, the AP will maintain the existing bank angle. ROLL annunciates on the PFD when the AP is engaged and no lateral FD mode is selected.

If no FD pitch modes are selected on the coupled side with the AP engaged, the AP will maintain the existing pitch attitude, and PITCH will annunciate on the PFD.



YD

The yaw damper automatically engages when the AP is engaged, but it may be deselected by depressing the YD selector. The yaw damper may be selected when the autopilot is not engaged by depressing the YD button. YD is described further in Chapter 15—"Flight Controls."



TURB

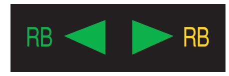
The turbulence mode of AP operation may be alternately selected and cleared by depressing the TURB button. When TURB mode is selected, the FCC adapts autopilot servo gains for turbulent flight conditions. Turbulence mode automatically clears when the AP is disengaged or at LOC capture.

TRIM

The red TRIM annunciator, in the center of the AP control panel, illuminates when the FCCs detect pitch trim failure. The autopilot should be disengaged in this case and not used. See Autopilot Trim Light Illuminates In Flight procedure in the Abnormal Procedures section of the *AFM*.

Mistrim

If the aircraft is out of trim while the AP is engaged, a boxed amber E (elevator), A (aileron), or R (rudder) will annunciate on the PFDs. If the mistrim condition indication continues for an extended period, the autopilot should be disconnected and the condition corrected. See Autopilot Mistrim Light Illuminated In Flight procedure in the Abnormal Procedures section of the *AFM*.



RB

The green RB illuminates when the rudder boost is active and the amber RB illuminates when rudder boost is not armed (RUDDER BOOST switch or avionics masters OFF) or when the FCCs detect a rudder boost failure. The rudder boost system is described further in the Flight Controls chapter.

AP Disengage

There are six different operator actions that will cause the AP to disengage. They are listed in descending order with the most commonly used method first to the least likely used.

- Depressing the arming button on either trim switch and moving the barrel in any of the four trim directions (if in primary pitch trim).
- Depressing the AP selector button on the AP control panel.
- Depressing the GO-AROUND button on the left thrust lever. This is the preferred method for go-around from a coupled approach.
- Depressing either wheel master switch (MSW). This method also disengages the YD.
- Placing the pitch trim selector switch to the OFF position.
- Moving the spring-loaded secondary trim switch to NOSE UP or NOSE DOWN (if in secondary pitch trim).

Whenever the operator disengages the AP, a disengage tone will sound, and the indicator light above the AP selector button will flash for five seconds and then extinguish. Also, the green AP → or AP ← annunciators on the PFDs will turn yellow and flash for five seconds, then turn white and stop flashing.

Automatic disengagement occurs anytime a failure condition is detected by FCC moni-

toring of the AP. If the monitor system disengages the AP, the disengage tone sounds, and the AP→ or AP ← annunciators flash until acknowledged by the pilot. The pilot acknowledges by activating AP disconnect (trim switch, MSW, or GA) or reengaging the AP.

AP Manual Control

When the AP is in PITCH or ROLL hold mode, pitch or roll commands can be made to the AP through the pitch/roll command switch barrel on the on-side control wheel. This is the same switch that is normally used to make primary pitch and aileron trim inputs while hand-flying the aircraft, however, when using it to command AP pitch and roll, the arming button is not depressed. Moving the barrel of this switch forward or aft, without depressing the arming button, will command pitch changes through the AP. If the barrel of the switch is moved left or right without depressing the arming button, roll will be commanded. Depressing the arming button and moving the barrel in any direction will disengage the AP as described above.

When the pitch/roll command switch is used to command roll, any FD lateral modes that where captured (except APPR and LOC) will be cleared and the AP will maintain the roll attitude existing when the pitch/roll command switch is released. When a pitch change is commanded, any FD pitch modes that where captured (except glideslope) will be cleared and the AP will maintain the existing pitch attitude when the pitch/roll command switch is released.

The pitch/roll command switch will not command pitch change when glideslope is captured. Also, roll cannot be commanded when approach or localizer is captured.

FLIGHT DIRECTOR CONTROLS AND OPERATIONS

General

The pilot and copilot mode selector panels (MSPs) are identical and completely independent of each other. Flight director (FD) modes are selected with ten push on/push off buttons on each MSP. When a mode is selected, incompatible modes automatically clear. A vertical line on the MSP separates the lateral and vertical modes.

The following description of the FD mode select panel applies to both the pilot and copilot mode select panels since they operate identically. Lateral modes are ROLL, HDG, 1/2 BANK, NAV, and APPR. Vertical modes are PITCH, ALTS (altitude preselect), ALT, LVL CHG (climb or descend), VNAV, VS, and SPD (IAS or Mach). Go-around is both a lateral and a vertical mode. Note that the roll, pitch, climb, descend, IAS, and Mach modes are not labeled mode buttons. The roll and pitch modes are attained by not having any other FD mode selected in that axis. The LVL CHG and SPD buttons are the switches used to attain the other modes, but they operate differently than simple mode select buttons. They are described later in this section.

In addition to the mode select buttons on the AP control panel and FD mode select panels, switches on the control wheel provide selection of vertical (pitch) sync (SYNC) and clearing of the flight director modes (FD CLEAR). A switch on the left throttle provides selection of go-around mode (GA). These are covered later in this section.

FD Mode Annunciation

When a mode button is depressed, and the FCC determines conditions are acceptable, the green indicator above the mode select button illuminates.

The FD modes annunciate on three lines above the top left corner of the on-side PFD attitude display. Lateral modes display above vertical modes. Active modes display in green and annunciate to the left of a blue vertical divider line. Armed modes display in white and annunciate to the right of the blue vertical line.

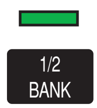
FD Lateral Modes

ROLL (Hold)

Roll mode is the basic lateral operating mode and occurs automatically when the V-bars are in view and no other lateral mode is active. ROLL annunciates on the PFD.

HDG (Heading)

Pressing the HDG button alternately selects or deselects the heading mode. Commands are generated to capture and maintain the selected heading, indicated by the heading bug on the PFD and MFD compass displays. The selected heading may be changed with the HDG knob on the CHP.



1/2 BANK

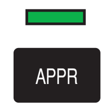
Pressing the 1/2 BANK button alternately selects or deselects half-bank mode. This mode reduces the maximum bank angle command to half the normal value. This mode automatically engages at 41,500 feet or above with any other lateral mode and automatically clears when the aircraft descends below 41,500 feet.

NAV (Navigation)

Pressing the NAV button alternately selects or clears the navigation mode. The FCC generates lateral commands to fly the active navigation course. The (active course) NAV identifier annunciates on the PFD (FMS, VOR 1, LOC 2, etc.). FMS, VOR or LOC will appear to the right of the vertical line on the PFD when the mode is armed and will be white. When the mode captures, the identifier moves to the left of the vertical line and turns to green. The NAV mode arms when the button is depressed, and automatically captures when capture conditions are met. Before capture, the system operates in the currently active lateral mode.

If FMS is the active NAV source, the capture point is determined by the FMC. After capture, the FMC applies lateral bank commands to the FCC.

If FMS is not the active NAV source, the FCC performs an all-angle adaptive capture. After capture, the FCC generates commands to maintain the VOR course or localizer beam.



APPR (Approach)

Pressing the APPR button alternately selects or clears the approach mode. The type of approach is determined by the active navigation source (selected on the CDU) and annunciates on the PFD the same as described for the NAV mode. APPR mode arms when the button is depressed, and automatically captures when capture conditions are met. Before capture, the system operates in the currently active lateral mode.

In an FMS approach, the capture point is determined by the FMC. After capture, the FMC applies lateral bank commands to the FCC.

In a non-FMS approach, the FCC performs an all-angle adaptive capture. The FCC arms for glideslope capture (if GS valid) after a front course localizer capture. GS annunciates in white to the right of the vertical line on the PFD. After glideslope capture, GS annunciates in green to the left of the blue vertical line, and the FCC generates commands to maintain the glidepath. All other vertical modes are automatically cleared at GS capture.

Back course mode automatically selects during a localizer approach if the selected course differs from the heading by more than 105°. BACK CRS annunciates on the PFD/MFD vertical deviation displays.

FD Vertical Modes

PITCH (Hold)

Pitch mode is the basic vertical operating mode and occurs automatically when the V-bars are in view and no other vertical mode is active.

ALTS (Altitude Preselect)

Depressing the ALTS button alternately arms or clears the altitude preselect mode. ALTS will arm right of the blue vertical line when the altitude preselect knob is rotated, a vertical mode is selected, or the SYNC button is depressed after a GA.

ALTS capture occurs when the aircraft approaches the preselected altitude and is dependent upon the rate of closure to that altitude. ALTS CAP will annunciate and flash left of the blue vertical line, indicating the altitude is being captured. When the new altitude is being tracked, the CAP will drop off and annunciate ALTS flashing for about three seconds and revert to steady. The aircraft is now in the ALTS track or hold.

WARNING

During the annunciation of an ALTS CAP, no other vertical mode should be selected, or the altitude capture sequence will be cancelled without re-arming the altitude preselect mode. This commonly occurs when the Pitch barrel is engaged during the ALTS CAP sequence. However, if the altitude preselect knob is rotated during the ALTS capture sequence, the ALTS CAP will be aborted and the ALTS will re-arm.

After the altitude is being tracked, and it is desired to go to a different altitude, the new altitude will be set using the altitude preselect knob. At this point, ALTS white will annunciate, right of the blue vertical line, and ALT green will annunciate left of the blue vertical line. The aircraft will remain at the current altitude until a vertical mode is selected. Once the new vertical mode is selected, it will annunciate green, left of the blue line.

ALT (Altitude Hold)

Pressing the ALT button alternately selects or clears the altitude hold mode. ALT annunciates in green on the PFD. The FCC generates commands to maintain the pressure altitude existing when ALT mode is selected.

Altitude hold mode automatically selects if the preselect altitude setting is changed (on ARP) while in altitude preselect track.

LVL CHG (Level Change)

Press the LVL CHG button to select a customized level change profile toward the preselected altitude (set on ARP). Either climb mode or descend mode results. Climb mode automatically selects if the aircraft altitude is below the preselected altitude. Descend mode selects if the aircraft altitude is above the preselected altitude.

LVL CHG (Climb Mode)

When the aircraft is below the preselected altitude, press the LVL CHG button to select the normal climb profile. Press the LVL CHG button a second time to select the high-speed climb profile. Pressing the LVL CHG button a third time clears the climb mode. Figure 16-32 depicts the normal and high-speed climb mode profiles.

When climb mode is selected, the FCC generates commands to fly either an IAS climb profile or a Mach climb profile (or a default minimum climb rate of zero—no descend will

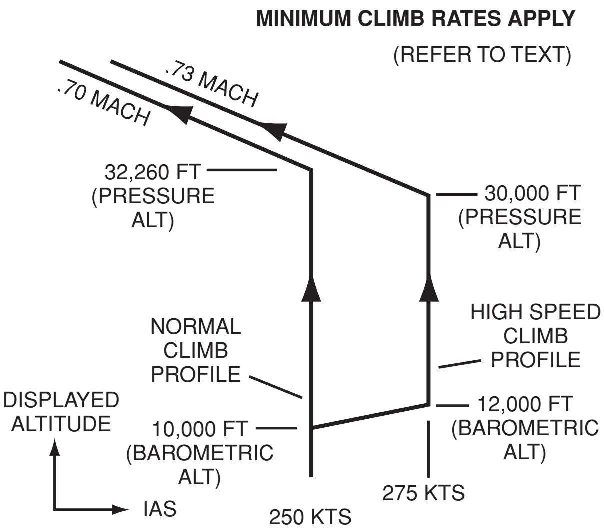


Figure 16-32. Climb Profiles

be commanded). An IAS profile is used at lower altitudes, and a Mach profile is used at higher altitudes. An IAS to Mach transition occurs automatically within the normal climb mode. CLM and either the IAS profile value (XXX) or the Mach profile value (.XX) annunciates on the PFD. An H also annunciates when the high-speed climb profile is selected.

LVL CHG (Descend Mode)

When the aircraft is above the preselected altitude, press the LVL CHG button to select a descend profile mode. Either an IAS/Mach or a VS based mode results. An IAS/Mach based mode selects if the aircraft is above 8,000 feet. A VS based mode selects if the aircraft is below 8,000 feet (Figure 16-33).

Above 8,000 feet

Press the LVL CHG button to select the normal (IAS/Mach) descend profile. Press the LVL CHG button a second time to select the high-speed descend profile. Pressing the LVL CHG button a third time will clear the descend mode.

When descend mode is selected, the FCC generates commands to fly either an IAS or a Mach descend profile (or a default minimum descend rate of 250 FPM). The IAS profile is used at lower altitudes and the Mach profile is used at higher altitudes. A Mach to IAS transition occurs automatically within the normal descend mode. Both normal and high speed profiles will slow the aircraft to 250 knots by 10,000 feet.

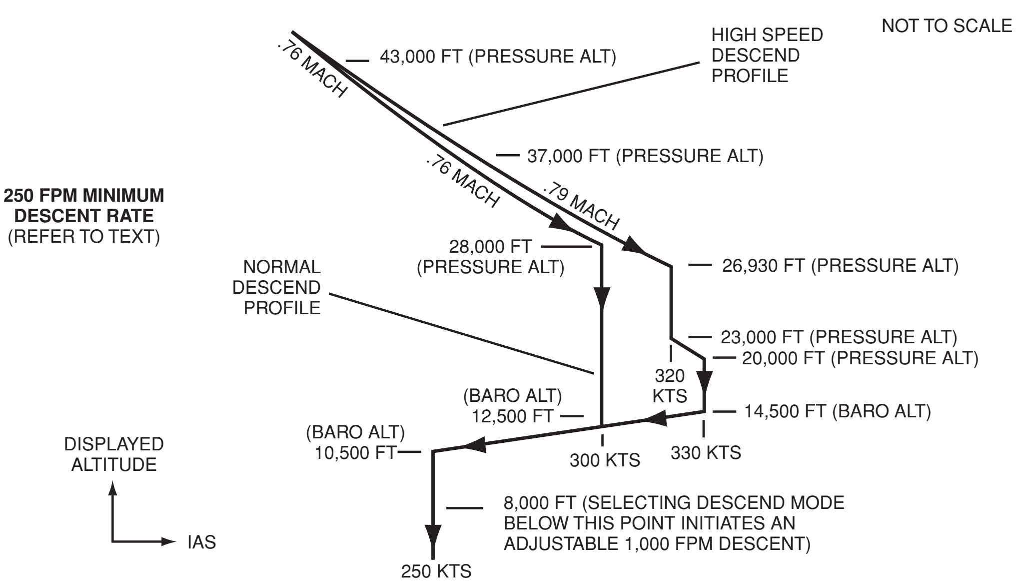


Figure 16-33. Descend Profiles

DES and either the IAS profile value (XXX) or the Mach profile value (.XX) annunciates on the PFD. An H also annunciates when the high-speed descend profile is selected.

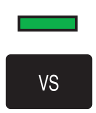
Below 8,000 feet

Press the LVL CHG button to select the (VS) descend profile mode. Press the LVL CHG button a second time to clear the descend mode.

DES and the VS profile value (DES 1.0 ↓) annunciates on the PFD. When descend mode is selected, the FCC generates commands to fly the vertical speed descend profile (1,000 FPM). Rotate the VS reference knob on the ARP to change the vertical speed profile value.

VNAV (Vertical Navigation)

Press the VNAV button to alternately arm or clear the vertical navigation mode. VNV annunciates in white to the right of the blue vertical line on the PFD. The FMC determines the VNAV capture point. After capture (VNV annunciates in green), the FMC applies vertical steering commands to the FCC. The VNAV mode automatically cancels when the vertical waypoint is reached. Refer to the *Universal Pilot's Guide* for more information on VNAV operation.



VS (Vertical Speed)

Press the VS button to alternately select or clear the vertical speed mode. VS and the vertical speed reference value annunciates on the PFD. An up arrow also annunciates for positive VS; a down arrow annunciates for negative VS. The FCC generates commands to maintain the vertical speed existing when VS mode is selected. Rotate the VS reference knob on the ARP to change the vertical speed reference value.

SPD (Speed)

Press the SPD button to select a speed hold mode. Either IAS mode or Mach mode selects depending on aircraft altitude at the time. IAS mode selects if the aircraft is below 30,000 feet. Mach mode selects if the aircraft is above 30,000 feet. Press the SPD button again to select the other (Mach/IAS) speed hold mode (transition is not automatic). Press the SPD button a third time to clear the speed hold mode.

Anytime an overspeed condition occurs, IAS or Mach mode will automatically select. Press the SPD button once to clear the speed hold mode. Refer to the IAS and Mach mode descriptions that follow. The SPD mode will not provide automatic 250 knots protection by 10,000 feet descending.

SPD (IAS Mode)

When the aircraft is below 30,000 feet, press the SPD button to select IAS hold mode. IAS and the airspeed reference value annunciate on the PFD. The FCC generates commands to maintain the airspeed existing when the IAS mode (SPD) is selected. Rotate the IAS knob on the ARP to change the airspeed reference value.

The IAS mode automatically selects if a significant overspeed (V_{MO} +10 KIAS) condition occurs and altitude is below 30,000 feet. Upon automatic (overspeed) selection of the IAS mode, any previously active vertical mode is cleared, and the IAS reference is set to V_{MO} –5 KIAS. When IAS mode has been selected automatically, it cannot be deselected until the IAS has decreased below V_{MO}.

SPD (Mach Mode)

When the aircraft is above 30,000 feet, press the SPD button to select the Mach hold mode. Mach and the Mach reference value annunciate on the PFD. The FCC generates commands to maintain the Mach speed existing when the Mach mode was selected. Rotate the IAS knob on the ARP to change the Mach reference value.

The Mach mode automatically selects if a significant overspeed (M_{MO} +.015 Mach) occurs and altitude is above 30,000 feet. Upon automatic (overspeed) selection of the Mach mode, any previously active vertical mode is cleared, and the Mach reference is set to M_{MO} –.015 Mach. When Mach mode has been selected automatically, it cannot be deselected until the Mach has decreased below M_{MO}.

Special Modes

SYNC Mode

The SYNC switch is the lower button on the inboard horn of each control wheel (Figure 16-34).

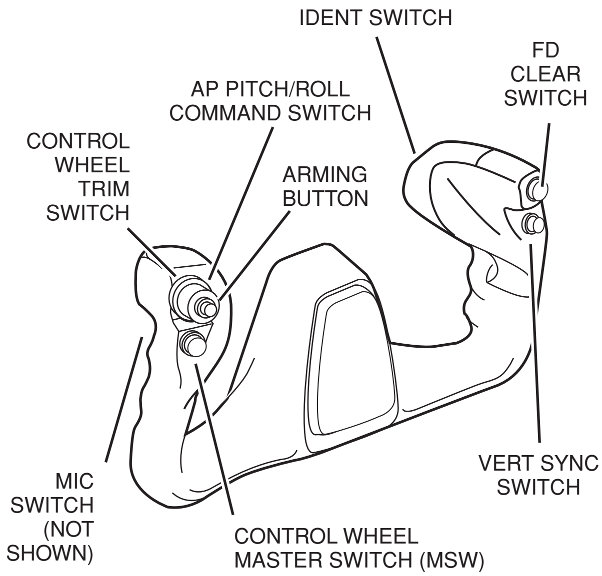


Figure 16-34. Pilot Control Wheel Switches

The FD SYNC switch is used, when not coupled to the AP, to synchronize (set) the vertical and lateral references to those currently being flown.

The vertical references that may be synchronized when not coupled are IAS (if in IAS mode), Mach (if in Mach mode), VS (if in VS mode), altitude hold memory (if in altitude hold mode), and the pitch angle memory (if in pitch mode). Go-around (vertical mode portion) is cleared by SYNC switch operation. Overspeed, vertical capture modes (ALTS, GS, LVL CHG speed modes, and VNAV), and profile modes are not affected by sync switch operation. The only lateral modes that can be synced are the bank and heading memories of roll mode.

FD CLEAR (Flight Director Clear)

The FD CLEAR switch is the upper button on the inboard horn of each yoke (Figure 16-34). The FD CLEAR switch is used to remove FD steering and mode information from the PFD when not coupled to the AP. When coupled to the AP, the FD CLEAR switch on the coupled side is not functional.

GA (Go-Around)

The GA switch is located on the side of the left thrust lever knob. Depressing the GA button sets the FD command bars to a fixed 9° pitch position on both flight directors and clears all other FD modes.

When airborne, a command is generated in the lateral axis to hold the heading existing at the moment of go-around selection. When go-around is selected on the ground, the lateral axis generates a wings-level command until takeoff. After takeoff, the lateral axis generates commands to hold the heading that existed at the moment of takeoff.

Selection of go-around disengages the autopilot (not yaw damper) and clears all other modes. Conversely, coupling the AP clears go-around. Go-around is also cleared by sync switch operation or automatic capture of another vertical mode.

NOTE

GA is both a vertical (9° pitch) and lateral (heading memory) mode. GA will not clear 1/2 BNK, if previously selected.

INTENTIONALLY LEFT BLANK

QUESTIONS

1. Which radio(s) is/are operative with both avionics master switches off?
   A. VHF COMM 1 and VHF COMM 2
   B. VHF COMM 1 only
   C. None, since the RTUs are on the avionics masters
   D. VHF COMM 1, if the OFF-RTU switch on the pilot ECP is selected to OFF

2. What is the result if both pilots select AHS cross-side?
   A. Nothing, both will continue to get on-side AHS information.
   B. Both pilots will receive cross-side AHS information.
   C. The pilot will get cross-side AHS information; the copilot will not.
   D. Neither pilot will get AHS reversion.

3. If an ADC fails in flight, the red ADC L or R annunciator light will illuminate.
   A. True
   B. False

4. If a bird strikes and damages the upper right pitot-static probe, which instruments could be affected?
   A. Pilot, if on-side ADC selected.
   B. Pilot, if cross-side ADC selected.
   C. Copilot, if on-side ADC selected.
   D. Stby Mach/IAS and Stdby altimeter.

5. The AHSs both begin a 70 second initialization process,
   A. When DC electrical power is applied to the aircraft electrical system.
   B. When the avionics master switches are turned on.
   C. When the AHS switches are turned on.
   D. When AC and DC electrical power is applied to the aircraft electrical system.

6. Heading information to the SDU is provided by:
   A. AHS 2, unless the pilot has cross-side selected.
   B. It receives the same heading information selected for the pilot side PFD.
   C. Always by AHS 1.
   D. It is selectable on the SDU.

7. A yellow boxed PIT on the PFDs indicates what?
   A. The pilot AHS is in error by 4° or more in pitch.
   B. The copilot AHS is in error by 4° or more in pitch.
   C. There is a disagreement of 4˚ or more in pitch between the two AHSs.
   D. One of the two AHSs has failed.

8. The left RTU cannot be used to tune the radios if the OFF-RTU switch on the left ECP is selected to OFF.
   A. True
   B. False

9. Transponder altitude reporting is turned on and off with a RTU line key on the ATC mode page.
   A. True
   B. False

10. The left RTU and COM 1 continue to operate in the EMER BUS mode of operation.
    A. True
    B. False

11. What navigation instrument will be available during EMER BUS mode of operation?
    A. Pilot PFD and MFD
    B. Pilot PFD only
    C. Copilot PFD only
    D. SDU

12. With the SDU PWR 1 circuit breaker out, an ILS approach can still be flown using the SDU.

A. True
B. False

13. Failure of an AHS is indicated by what?

A. Boxed red ATT and MAG flags on the on-side PFD
B. Boxed yellow PIT and ROL flags and removal of heading and attitude information from the on-side PFD
C. Illumination of the red L or R AHS annunciator
D. Box red HDG flag on the on-side PFD

14. The brightness of the PFD and MFD displays is controlled:

A. Automatically
B. With individual brightness (BRT) knobs in the upper left corner of the units
C. With the EFIS dimmer knobs on the L and R INSTR LIGHTS panels
D. Either B or C

15. What does the low speed cue (red and black checkered bar) on the PFDs represent?

A. Computed shaker speed
B. 1.3 V_S
C. Aerodynamic stall
D. Computed pusher speed

16. Setting the IAS bug value on one PFD always changes the bug value on both PFDs.

A. True
B. False

17. If a CDU fails and cross-side is selected on the ECP, DH/RPT/MDA are all controlled with the cross-side AAP.

A. True
B. False

18. How is the active NAV source selected?

A. It is selected on the MFD using the NAV SRC page.
B. It is selected on the ECP using the CDU button.
C. On aircraft with Collins FMS, it is selected on the CDU using the NAV SRC key and active NAV source menu page.
D. On aircraft with UNS FMS, it is selected on the ERP using the NAV key and the NAV SOURCE menu page on the CDU.

19. The BRG key on the UNS ERP is used to select the navaid source for the bearing pointers on the PFDs, MFDs and SDU.

A. True
B. False

20. When the red TRIM annunciator illuminates on the AP control panel:

A. The AP should be disengaged, aircraft retrimmed, and then AP may be reengaged.
B. The AP should be disengaged and not used.
C. The AP pitch axis should be disengaged but the roll axis may remain engaged.
D. The AP will automatically disengage.

CHAPTER 17
MISCELLANEOUS SYSTEMS

CONTENTS

ILLUSTRATIONS

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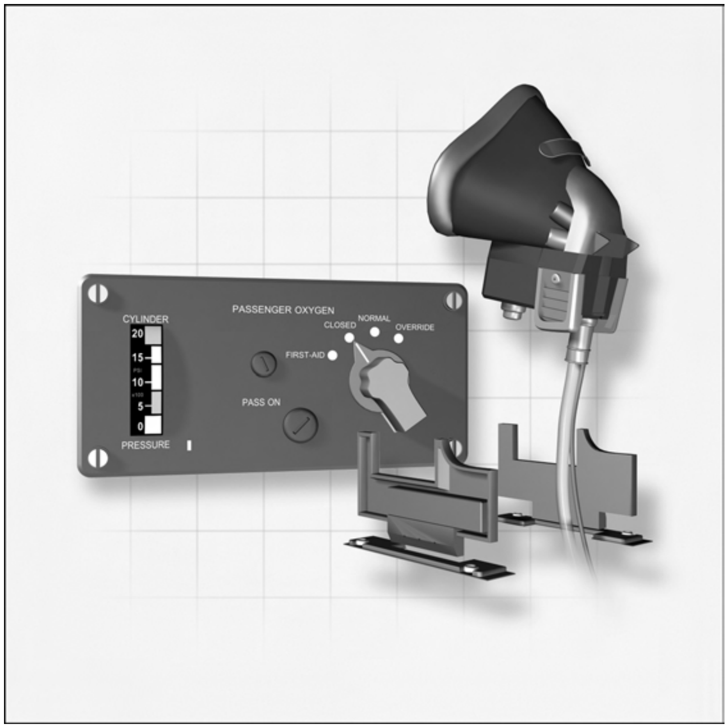
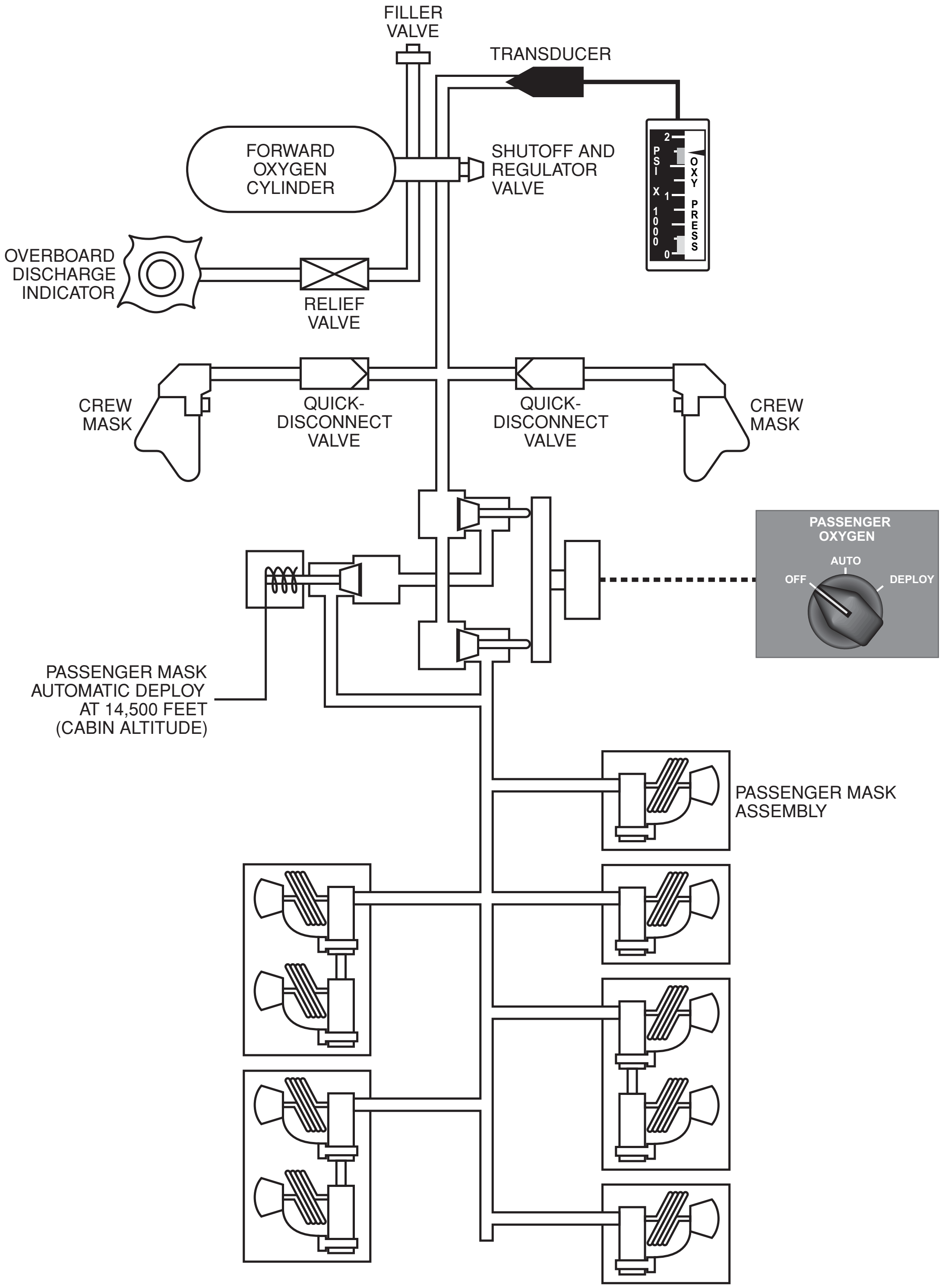
CHAPTER 17
MISCELLANEOUS SYSTEMS



OXYGEN SYSTEM

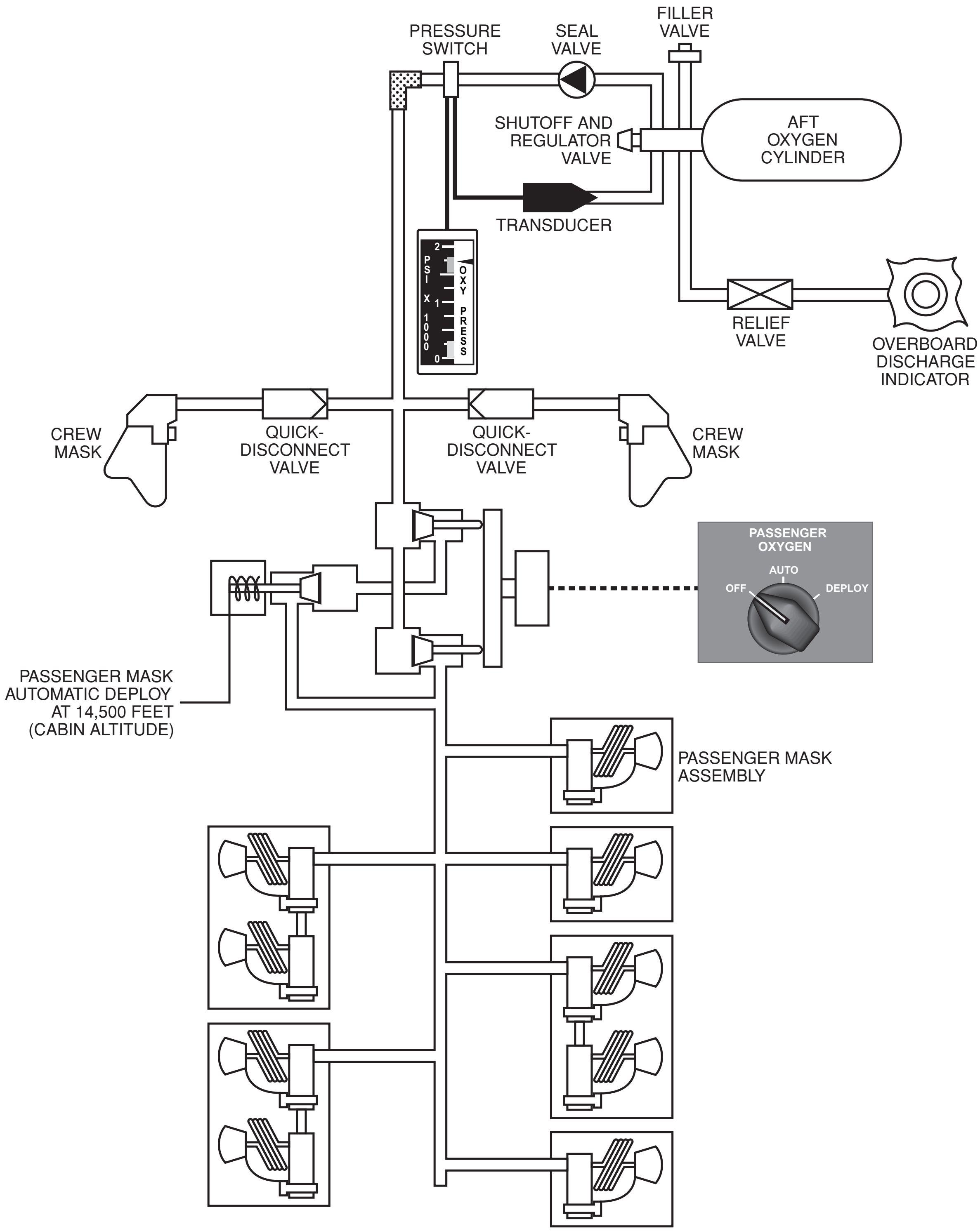
GENERAL

The aircraft oxygen system (Figure 17-1) provides oxygen service for the crew and passengers. The system consists of the crew and passenger distribution systems, a high-pressure oxygen storage cylinder, a shutoff valve and pressure regulator assembly, an oxygen pressure transducer, an oxygen pressure indicator, an overboard discharge relief valve and indicator, a passenger oxygen control valve, lanyard actuated passenger mask oxygen valves, and crew and passenger oxygen masks. Electrical power to operate the passenger oxygen control valve and oxygen indicator is supplied through the 7.5-amp OXYGEN VALVE circuit breaker on the pilot circuit breaker panel. Oxygen is available to the crew at all times and can be made available to the passengers either automatically above 14,500 feet cabin altitude, or manually, at all altitudes through the use of the cockpit controls on the pilot circuit breaker panel. The oxygen system is designed for use during emergency descent to a cabin altitude not requiring oxygen and is not to be used for extended periods of flight at cabin altitudes requiring oxygen or as a substitute for the normal pressurization system. Smoking is prohibited when oxygen is in use.



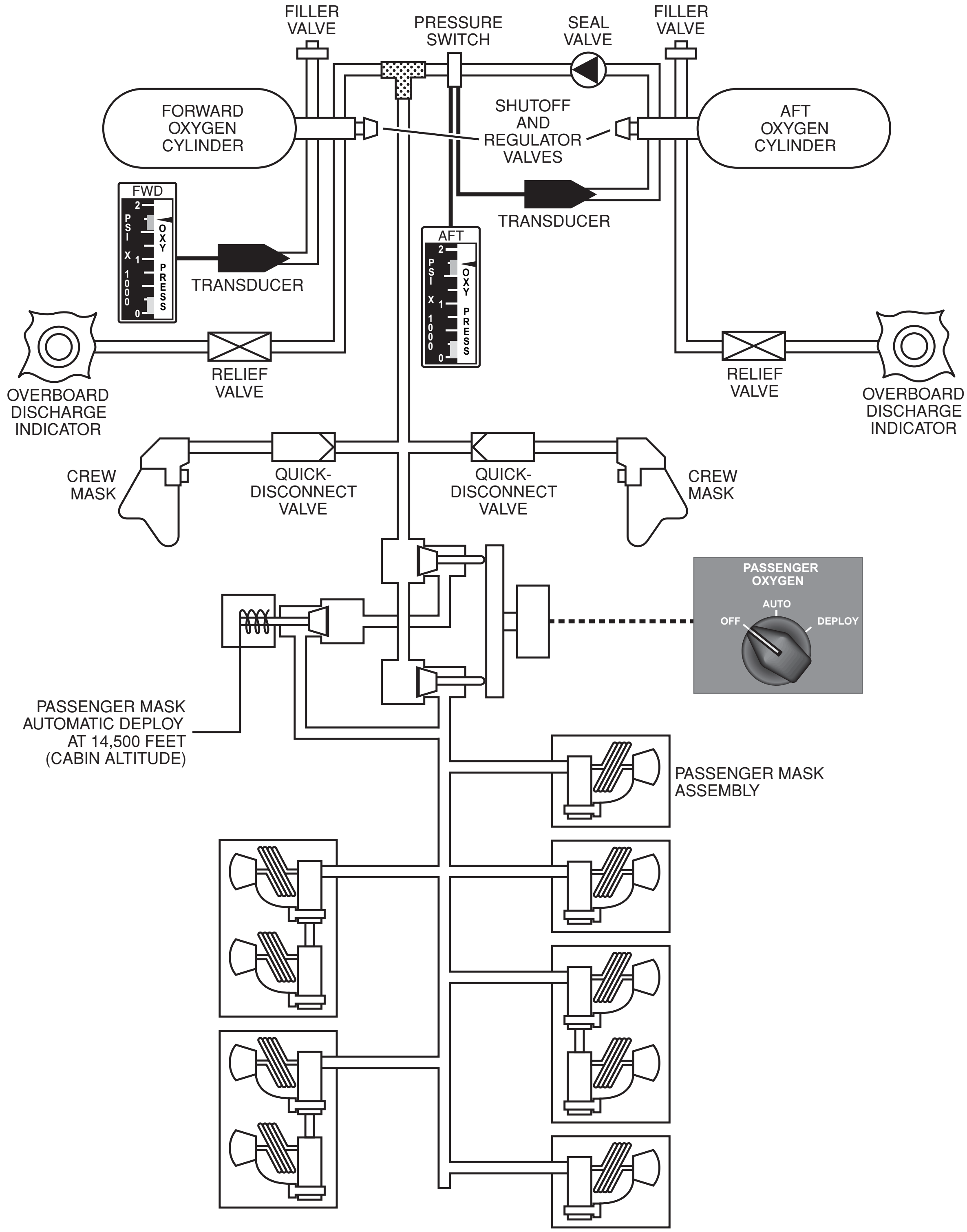
WITH SINGLE FORWWARD CYLINDER

Figure 17-1. Oxygen System Diagram (Sheet 1 of 3)



WITH SINGLE AFT CYLINDER

Figure 17-1. Oxygen System Diagram (Sheet 2 of 3)



WITH DUAL CYLINDERS

Figure 17-1. Oxygen System Diagram (Sheet 3 of 3)

OXYGEN STORAGE CYLINDER

The system is supplied with oxygen by a storage cylinder (Figure 17-2) located in the left aft nose avionics compartment or in the vertical stabilizer (or both locations). The oxygen storage cylinder has a storage capacity of 40 or 77 cubic feet at 1,800 psi. The following oxygen storage cylinder arangements are used:

- Single cylinder in the nose compartment (40 cubic feet)
- Single cylinder in the nose compartment (77 cubic feet)
- Single cylinder in the verticle stabilizer (77 cubic feet)
- Dual cylinders—One in the nose compartment (40 cubic feet) and one in the vetical stabilizer (77 cubic feet)
- Dual—One in the nose compartment (77 cubic feet) and one in the vertical stabilizer (77 cubic feet)

The shutoff and pressure regulator assembly forms an integral part of the storage cylinder and provides for pressure regulation, pressure indication, and servicing. Oxygen pressure for the passenger and crew distribution system is regulated at 60 to 80 psi. Under normal conditions, this valve should always be left in the open position. The pilot should be aware that on aircraft 60-001 to 60-103 the oxygen cylinder shutoff valve can be closed and oxygen pressure still can be read on the OXY PRESS indicator in the cockpit. On aircraft 60-104 and subsequent, a pressure switch (Figure 17-1) has been added to the oxygen feed line. On these aircraft, the cockpit gage will show line pressure and read zero if the oxygen cylinder shut off valve is closed or the line is blocked. During preflight, ensure that the system is activated by checking for airflow through a crew oxygen mask.

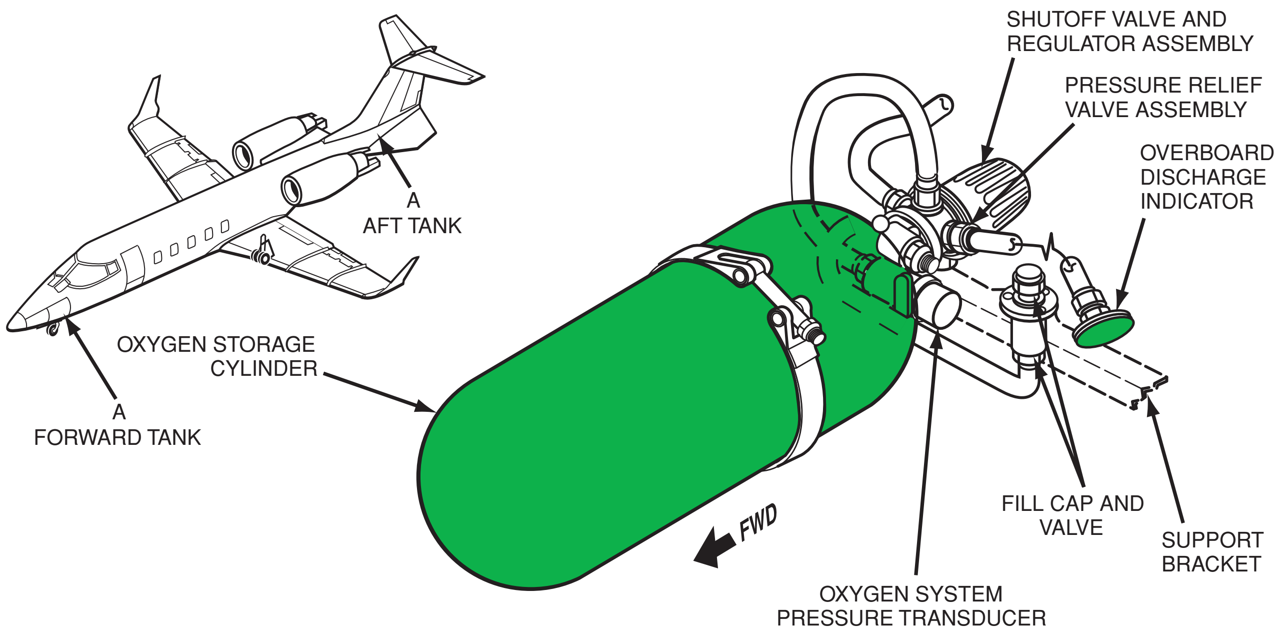
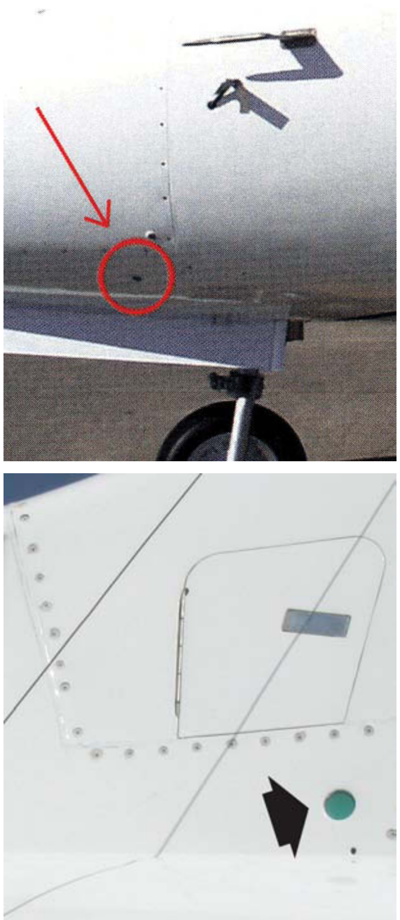


Figure 17-2. Oxygen Storage Cylinder

OVERBOARD DISCHARGE INDICATOR

The overboard discharge indicator (green blowout disc) (Figure 17-3) provides the pilot with a visual indication that there has not been an overpressure condition in the oxygen storage cylinder. The burst disc pressure relief valve blows out at 2,700 to 3,000 psi, releasing all oxygen pressure. System pressure should normally be between 1,550 and 1,850 psi. The green blowout disc is located on the lower left side of the nose section or on the left aft fuselage below the vertical stabilizer.



FORWARD TANK

AFT TANK

Figure 17-3. Overboard Discharge Indicator

OXYGEN PRESSURE INDICATOR

The OXY PRESS indicator and passenger oxygen control knob (Figure 17-4) provides a readout of the amount of oxygen available in the cylinder. The indicator uses a vertical-type display and is located on the pilot sidewall panel. The indicator face has a green segment between 1,550 and 1,850 psi and has a red line at 2,000 psi. The oxygen supply system may be a single cylinder or a dual cylinder system. In aircraft with dual systems, a second pressure indicator is added to the pilot circuit breaker panel to allow determination of the oxygen pressure in each cylinder. The transducer for the aft oxygen system is wired through a pressure switch in the aft pressure indicator. The pressure switch senses loss of pressure in the aft oxygen tube. If the aft cylinder is pressurized but the supply tube is not (for example; due to blockage), the indicator will read zero. Since pressure will vary due to temperature, the fore and aft cylinder may not indicate the same during flight. Electrical power to operate the oxygen indicator is 28 VDC supplied through the OXY VALVE circuit breaker in the pilot ENVIRONMENT circuit breaker group.

CREW DISTRIBUTION SYSTEM

There are two different flight crew oxygen masks available for the Learjet 60 aircraft: the Scott–ATO and the Puritan Bennett Sweep–On 2000.

CREW MASKS—SCOTT ATO

The flight crew oxygen masks (Figures 17-5 and 17-7) are stowed in accessible stowage boxes just aft of the pilot and copilot circuit breaker panels or in storage cups just aft of the pilot and copilot on the bulkhead. The mask regulators provide for normal, 100% oxygen, and emergency operation (refer to the *AFM* for detailed operational procedures). Each mask incorporates a microphone controlled by the NORM MIC/OXY MIC switch on the respective audio control panel (Figure 17-6). When

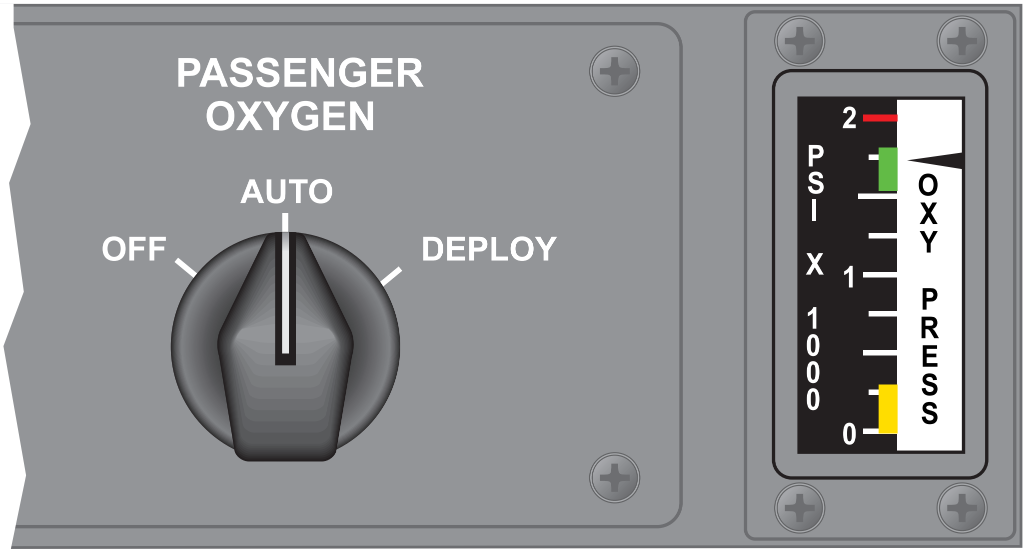


Figure 17-4. OXY PRESS Indicator and Passenger Oxygen Control Knob

Figure 17-5. Crew Oxygen Mask Storage Box

Figure 17-6. Pilot Audio Control Panel

the OXY MIC is in use, a voice-activated hot interphone exists for crew member communication. An optional oxygen pressure detector may be located in the oxygen line. If sufficent pressure is available in the line, the detector shows green.

CREW MASKS—PURITAN BENNETT SWEEP–ON 2000

The crew oxygen masks (Figure 17-7) are quick-donning masks that contain a mask mounted automatic diluter-demand regulator with a single knob mode control, an inflatable harness, and a microphone. The flight crew oxygen masks are stowed in a stowage cup just aft of the pilot and copilot circuit breaker panels. The mask regulators provide for normal, 100% oxygen, and emergency operation (refer to the FAA approved *AFM* for detailed operational procedures). The mask incorporates a microphone controlled by the NORM MIC/OXY MIC switch on the respective audio control panel (Figure 17-6). When the OXY

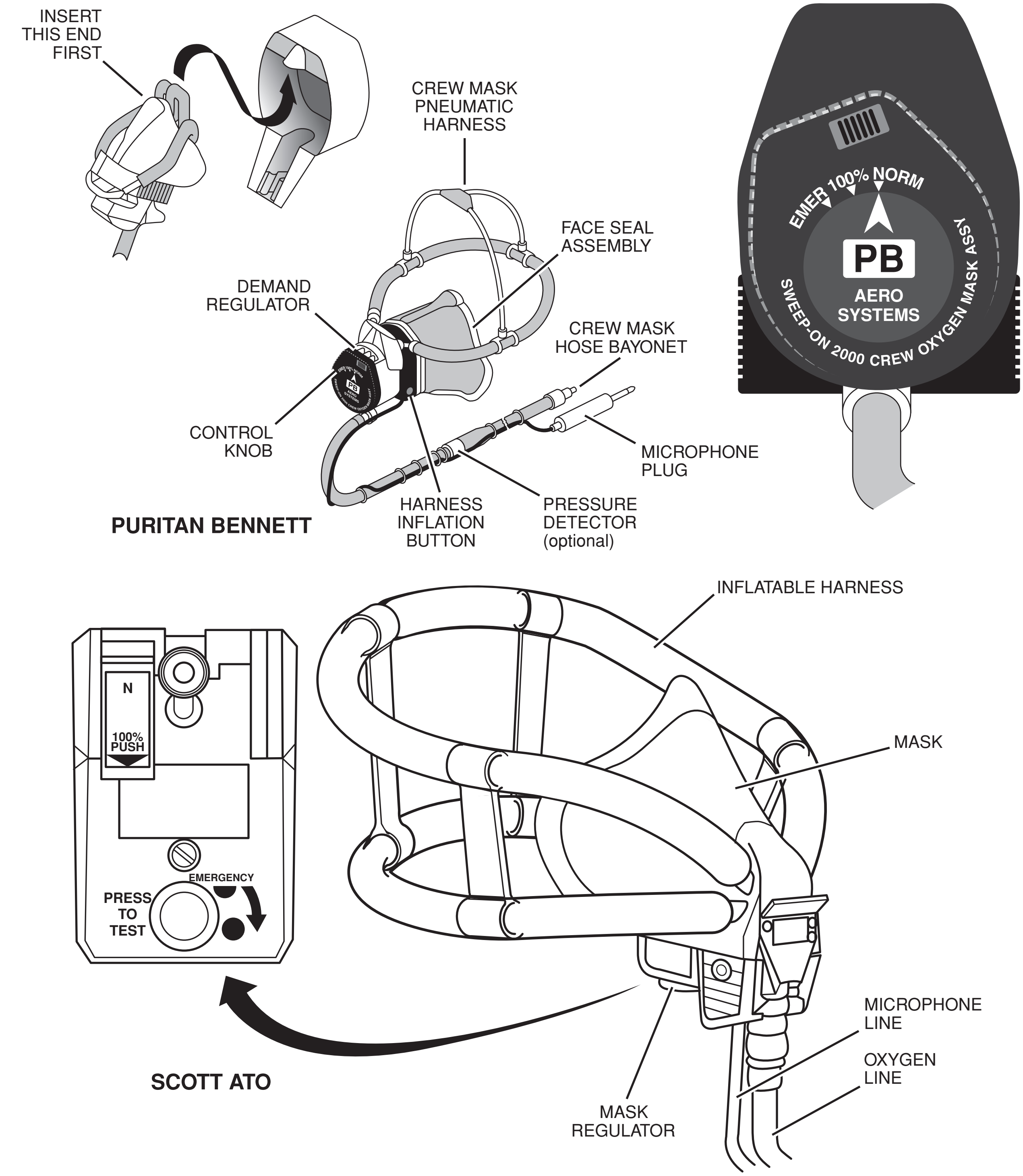


Figure 17-7. Crew Oxygen Mask

MIC is in use, a voice–activated hot interphone exists for crew member communication. An optional oxygen pressure detector may be located in the oxygen line. If sufficent pressure is available in the line, the detector shows green.

The storage box incorporates a PRESS-TO-TEST/RESET control and flow indicator (blinker) (see Figure 17-5). When the indicator shows Yellow, oxygen flow is present. When the indicator is black, no oxygen flow is present.

The PRESS-TO-TEST/RESET control on the storage box allows the mask to be tested while stowed. Depressing the PRESS-TO-TEST/RESET control will supply oxygen to the mask regulator. The blinker will turn yellow momentarily, then black indicating the oxygen shutoff and regulator valve is turned on and that the mask regulator is not leaking.

The mask regulator contains red handles that inflate the mask harness when squeezed, a red, up-down, 100% selector, and a rotating two-position EMERGENCY button/knob (Figure 17-7). The EMERGENCY knob also has a spring-loaded PRESS-TO-TEST function that allows the regulator demand system to be tested.

With the 100% lever extended (not selected) and the PRESS-TO-TEST button/knob rotated to the half-circle position, the mask will deliver automatic oxygen dilution from S.L. to 30,000 feet cabin altitude, 100% oxygen above 30,000 feet cabin altitude, and automatic pressure breathing above approximately 37,000 feet cabin altitude.

The wearer of the mask can select 100% oxygen at any time by depressing on the 100% lever. Pressure breathing can be selected at any altitude by rotating the EMERGENCY/PRESS-TO-TEST knob to the full-circle position. With the mask pressure regulator in this position, the mask will deliver 100% oxygen at all cabin altitudes and maintain a positive pressure in the mask cup at all times for respiratory protection from smoke or fumes.

CREW MASK OPERATION

The crew should be proficient at donning the oxygen masks quickly. There are four basic steps to donning the crew masks, as follows:

1. Remove any hat or earmuff-type headset in use. These items may interfere with the quick donning of the mask.

NOTE

Headsets and eyeglasses worn by crew members may interfere with quick-donning capabilities.

2. Grasp, squeeze, and hold the red handles on the mask's pressure regulator together. This action inflates the pneumatic harness for donning and releases the mask from the holder.
3. Position the harness on the head. Place the mask over nose and mouth, then release the handles. At this point, the elastic harness deflates, causing the mask to tighten about the head.
4. Ensure that the mask is properly sealed and reposition the mask as desired.

The mask should be stowed with the 100% lever depressed so it is ready to use in the event of smoke or fumes in the airplane; however, if the mask is used because of a pressurization problem, the 100% lever should be retracted if oxygen duration becomes a consideration. Refer to Table 17-1, Oxygen Duration Chart, in this section. The oxygen duration chart for the 77 cubic feet cylinder can be found in the Pilot Quick Reference Handbook, Supplemental Section.

The PRESS-TO-TEST knob must be in the half-circle position for mask stowage; however, once the mask is donned, if the diluter demand function doesn't appear to be working properly or if smoke/fumes are present, the emergency position (full-circle) should be selected.

Table 17-1. OXYGEN DURATION CHART (40 CUBIC FEET)

OXYGEN DURATION—MINUTES
FULLY CHARGED SYSTEM

FINAL STABILIZED CABIN ALTITUDE ~ 1,000 FEET	2 CREW	2 CREW 2 PASS	2 CREW 4 PASS	2 CREW 6 PASS	2 CREW 8 PASS	2 CREW 9 PASS	2 CREW 11 PASS
40	313 **313**	96 **96**	57 **57**	40 **40**	31 **31**	28 **28**	23 **23**
35	218 **218**	82 **82**	51 **51**	37 **37**	29 **29**	26 **26**	22 **22**
30	157 **157**	70 **70**	45 **45**	33 **33**	26 **26**	24 **24**	20 **20**
25	157 **114**	69 **59**	44 **40**	33 **30**	26 **24**	23 **22**	**20** **19**
20	209 **90**	76 **51**	47 **36**	34 **28**	26 **22**	24 **21**	20 **18**
15	279 **72**	82 **45**	48 **32**	34 **25**	26 **21**	24 **19**	20 **16**
10	279 **57**	**PASSENGER OXYGEN NOT REQUIRED**					
8	228 **52**						
7	208 **50**						

- BOLD FACE NUMBERS (XXX) INDICATE 100% OXYGEN.
- LIGHT FACE NUMBERS (XXX) INDICATE DILUTER DEMAND.
- CREW AND PASSENGER OXYGEN MASKS ARE NOT APPROVED FOR USE ABOVE 40,000 FEET CABIN ALTITUDE. **PASSENGER DURATIONS ABOVE 30,000 FEET CABIN ALTITUDE ARE PROVIDED FOR INFORMATION ONLY. PASSENGER MASKS WILL NOT PROVIDE SUFFICIENT OXYGEN FOR PROLONGED OPERATION ABOVE 34,000 FEET CABIN ALTITUDE. PROLONGED OPERATION ABOVE 25,000 FEET CABIN ALTITUDE WITH PASSENGERS ON BOARD IS NOT RECOMMENDED.**
- PRIOR TO OVERWATER FLIGHTS, PLAN OXYGEN REQUIREMENTS TO PROVIDE SUFFICIENT OXYGEN FOR ALL OCCUPANTS IN THE EVENT OF A PRESSURIZATION FAILURE. ADDITIONAL OXYGEN MAY BE REQUIRED TO ASSURE THAT BOTH OXYGEN DURATION AND RANGE (FUEL) REQUIREMENTS ARE SATISFIED.
- FOR CABIN ALTITUDES OF 10,000 FEET AND ABOVE, THE OXYGEN DURATION TIMES INCLUDE CABIN ALTITUDE ASCENT TIME FROM 8,000 FEET TO FINAL STABILIZED CABIN ALTITUDE.
- TO CALCULATE OXYGEN DURATION FOR A LESS THAN FULLY CHARGED SYSTEM THE FOLLOWING FORMULA MAY BE USED:

DURATION = DURATION FROM CHART X (SYSTEM PRESSURE ÷ 1,850)

PASSENGER DISTRIBUTION SYSTEM

The passenger distribution system (Figure 17-8) can be used to provide oxygen to the passengers in case of pressurization system failure or any other time that oxygen is required.

Oxygen is available in the crew oxygen distribution lines whenever the oxygen cylinder shutoff valve is open; however, oxygen is not admitted to the passenger distribution system until required. Oxygen supply to the passengers' system is controlled by a three-position (OFF-AUTO-DEPLOY) passenger oxygen knob located at the top of the pilot circuit-breaker panel (see Figure 17-4).

This knob controls two valves (See Figure 17-1). In addition to the two manually controlled valves, there is also a solenoid-operated valve which can be energized open by an

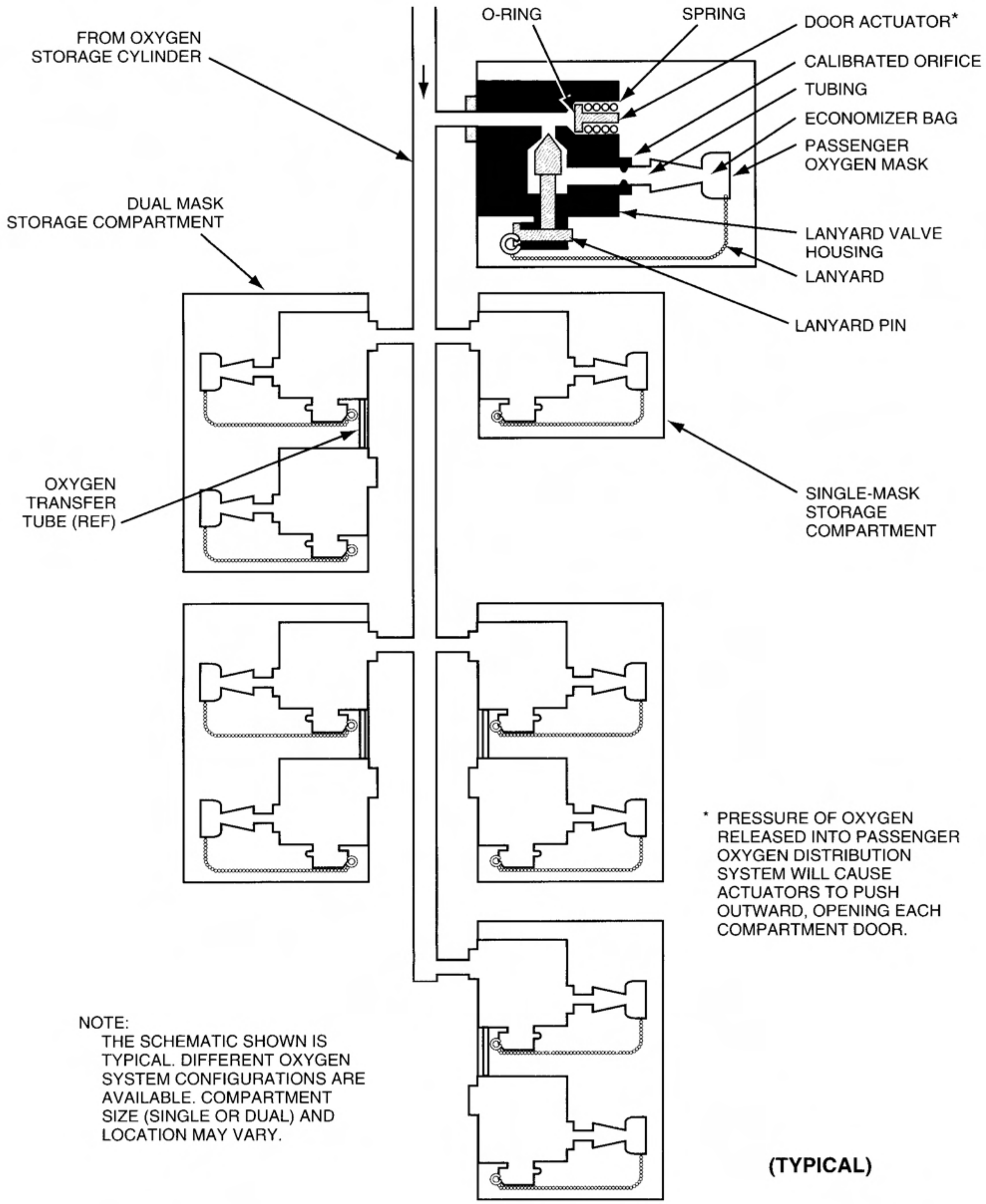


Figure 17-8. Passenger Distribution System Diagram

aneroid switch. The manually controlled PASSENGER OXYGEN knob is normally in the AUTO position, which makes oxygen pressure available through one of the manually controlled valves to the aneroid-controlled solenoid valve. Oxygen can be admitted to the passenger distribution system through either the solenoid valve (AUTO) or through the other manually controlled valve (DEPLOY).

When the PASSENGER OXYGEN valve is in the OFF position, oxygen will not be available to the passenger distribution system. This position may be used only when no passengers are being carried.

When the PASSENGER OXYGEN valve is in the AUTO position, oxygen will be automatically admitted to the passenger distribution system, through the aneroid-controlled solenoid valve, if the cabin reaches 14,500 ± 250 feet. The aneroid switch opens the solenoid-controlled valve and deploys the passenger masks. It also illuminates the cabin overhead lights.

In the event of aircraft electrical failure, automatic deployment of the passenger masks is not possible. The oxygen solenoid valve requires DC power through the OXY VALVE circuit breaker (pilot ENVIRONMENT circuit-breaker group) for automatic mask deployment.

Rotating the PASSENGER OXYGEN valve from AUTO to the DEPLOY position admits oxygen into the passenger distribution system immediately and causes the passenger oxygen masks to drop. This position can be used to deploy the passenger masks at any altitude, but will not cause the cabin overhead lights to illuminate.

The passenger oxygen masks (Figure 17-9) are stowed in compartments in the convenience panels above the passenger seats.

Some compartments will contain two masks, depending on the aircraft seating configuration. There will be at least one spare mask. Also, those aircraft configured with an enclosed lavatory will have two oxygen masks located in the lavatory.

The passenger mask storage compartment doors are normally held closed with latches. When oxygen is admitted to the passenger distribution system, the oxygen pressure will cause door actuators (plungers) to push outward, opening each compartment door. When the doors open, the passenger masks will fall free and be available for passenger donning. Each passenger must pull the lanyard attached to his or her mask to initiate oxygen flow. An orifice incorporated in the mask tubing will

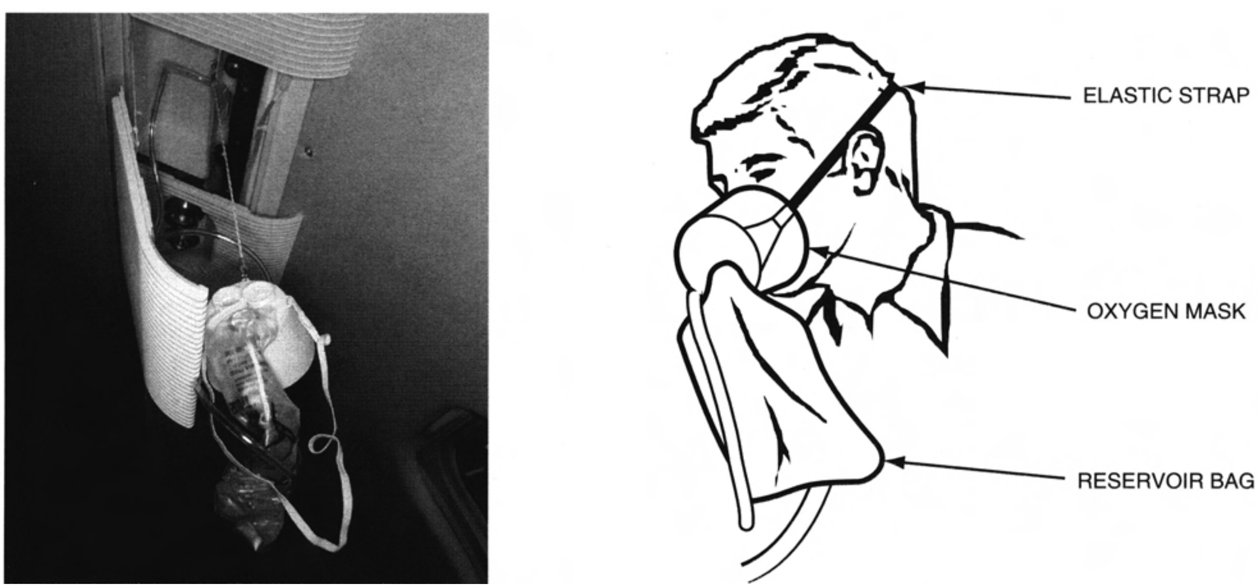


Figure 17-9. Passenger Mask

provide a constant flow rate of 4.5 liters per minute. The reservoir bag may seem to inflate slowly, but this is normal.

Should the doors be inadvertently opened from the cockpit, oxygen pressure must be bled from the passenger distribution system before the masks can be restowed. This is accomplished by pulling one of the forward passenger mask lanyards after ensuring that the PASSENGER OXYGEN valve is returned to the AUTO position.

The lanyard pin should be installed prior to closing the last door. If the doors open due to a malfunction of the solenoid-operated valve, the PASSENGER OXYGEN valve must be turned to the OFF position to permit stowage of the passenger masks.

The compartment doors can be opened manually for mask cleaning and servicing.

OXYGEN DURATION

Before an overwater flight is made, the pilot should plan oxygen requirements to provide sufficient oxygen for all occupants in case of a pressurization failure. Additional oxygen may be required to ensure that oxygen duration and fuel requirements are met. See *AFM* Section IV for Oxygen Duration chart.

The pilot should also remember that the crew and passenger masks are not approved for use above 40,000 feet cabin altitude; therefore, if a pressurization problem is encountered at a high altitude, a descent should be initiated immediately.

For cabin altitudes of 10,000 feet and above, the oxygen duration times listed in Table 17-1 include cabin altitude ascent time from 8,000 feet to the final stabilized cabin altitude.

The pilot may use the following formula to calculate the oxygen duration for a partially charged oxygen system:

Duration = Duration per Table 17-1

$$X \left(\frac{\text{System Pressure}}{1{,}850}\right)$$

Table 17-2 lists the average time of useful consciousness at various altitudes when the oxygen system is not used. See the Learjet 60 *AFM* sections 1, 2, 3, and 4 for Limitations, Crew Procedures— Normal, Emergency and Abnormal.

Table 17-2. AVERAGE TIME OF USEFUL CONSCIOUSNESS

51,000 feet	4 to 8 seconds
45,000 feet	9 to 15 seconds
40,000 feet	15 to 20 seconds
35,000 feet	1/2 to 1 minute
30,000 feet	1 to 2 minutes
28,000 feet	2 to 3 minutes
25,000 feet	3 to 5 minutes
22,000 feet	5 to 10 minutes
12,000 to 18,000 feet	30 minutes or more

INTENTIONALLY LEFT BLANK

QUESTIONS

1. During preflight, the pilot can determine if the oxygen bottle is turned on by:

A. Reading the pressure indicated on the oxygen pressure indicator in the cockpit

B. Ensuring that there is airflow through a crew mask

C. Placing the OXY/MIC switch to the OXY position

D. Reading the pressure indicated on the oxygen pressure indicator on the storage bottle

2. The crew checks for the availability of oxygen to the stowed crew masks by:

A. Depressing the PRESS-TO-TEST/RESET control on the storage box

B. Rotating the PRESS-TO-TEST button to the half-circle position

C. Depressing the 100% lever

D. Placing the OXY/MIC switch to the OXY position

3. Selecting the passenger oxygen knob to the DEPLOY position will:

A. Cause passenger masks to drop and turn on the cabin overhead lights

B. Prevent oxygen from entering the passenger oxygen distribution lines

C. Deploy the passenger masks but not deliver oxygen to the masks

D. Deploy passenger masks and release oxygen flow to the passenger masks

4. Which of the following statements is true if the passenger oxygen knob is in the AUTO position and electrical power is available?

A. Oxygen is supplied to the passenger masks if the cabin altitude reaches 10,000 feet.

B. Passenger masks will automatically deploy in the event of electrical failure.

C. Passenger masks will automatically deploy if cabin altitude reaches 14,500 feet.

D. B and C

5. The half-circle (◒) position on the crew mask PRESS-TO-TEST knob provides what (100% lever not depressed)?

A. 100% oxygen at all times

B. Diluter demand from S.L. to 30,000 feet cabin altitude, 100% oxygen above 30,000 feet, and pressure breathing above 37,000 feet

C. 100% oxygen at all cabin altitudes and pressure breathing at all times

D. Closure of the oxygen mask regulator to prevent dust from entering the valve

MANEUVERS AND PROCEDURES

CONTENTS

ILLUSTRATIONS

TABLE

CHAPTER 18
MANEUVERS AND PROCEDURES

INTRODUCTION

The general pilot information in this chapter is intended to supplement and expand upon information in other sources. It is not intended to supersede any official publication. If there is any conflict between the information in this chapter and that in any official publication, the information in the official publication will be used.

GENERAL

General pilot information includes standard operating procedures and the maneuvers normally encountered during Learjet training and operations. The following abbreviations are used in this chapter.

ABBREVIATIONS

AFM	*Airplane Flight Manual*
AGL	Above Ground Level
ATA	Airport Traffic Area
ATC	Air Traffic Control
CDI	Course Deviation Indicator
COM/NAV	Communication/Navigation
DH	Decision Height
FAF	Final Approach Fix
FL	Flight Level
HAA	Height Above Airport
HAT	Height Above Touchdown
IAF	Initial Approach Fix
KIAS	Knots, Indicated Airspeed
MAP	Missed Approach Point
MDA	Minimum Descent Altitude
M_{MO}	Mach, Maximum Operational
MSL	Mean Sea Level
N_1	Fan Speed
PF	Pilot Flying
PIC	Pilot In Command
PM	Pilot Monitoring
SOP	Standard Operating Procedure
VDP	Visual Descent Point
V_{FE}	Velocity Flaps Extended
V_{MO}	Velocity Maximum Operational
V_1	Takeoff Decision Speed
V_R	Rotation Speed
V_{REF}	Reference Speed
V_2	Takeoff Safety Speed

STANDARD OPERATING PROCEDURES

GENERAL

Standard operating procedures (SOPs) are used to supplement the information in the *AFM* and federal air regulations. Adherence to SOPs enhances individual and crew situational awareness and performance. SOPs may include assignment of responsibilities, briefing guides, and procedures to be followed during specific segments of flight. The SOPs in this chapter are not intended to be mandatory or to supersede any individual company SOPs. They are simply provided as examples of good operating practices.

RESPONSIBILITIES

PIC — The pilot in command is designated by the company for flights requiring more than one pilot, and is responsible for conduct and safety of the flight.

PF — The pilot flying controls the airplane with respect to heading, altitude, and airspeed and accomplishes other tasks as directed by the PIC.

PM — The pilot monitoring maintains ATC communications, obtains clearances, accomplishes checklists, makes altitude callouts and other tasks as directed by the PIC.

SIC — The second in command is the second pilot on an aircraft requiring more than one pilot for the flight.

The SIC is responsible for providing advice and counsel to the PIC. The PIC may choose to accept or reject such advice. That is a prerogative of the PIC. But neither the PIC's acceptance nor rejection of advice relieves the SIC of the responsibility for providing it.

CHECKLIST PROCEDURES

Normally, the PF will initiate all checklists. However, if the PM thinks a checklist should be accomplished and the PF has not called for it, the PM should prompt the PF. For example, "Ready for the Approach checklist, Captain?"

FlightSafety International recommends the use of the checklist challenge and response concept. Using Normal Procedures checklists, the PM will challenge the PF and the PF will respond. Using Abnormal or Emergency Procedures checklists, the PM will challenge the PF and, as a memory aid, will also give the checklist item response. The PF will then respond.

The PF may elect to have the PM accomplish some Abnormal or Emergency procedure checklists on the PF's command. In this case, the PM will give the checklist item and response. The PF will reply with the response and the PM will accomplish the action.

When a checklist has been completed, the PM will report the checklist is complete and that he is standing by with the next checklist. For example, "Approach checklist complete. Standing by with the Before Landing checklist."

If an emergency occurs on takeoff after V_1 speed and the takeoff is continued, no checklist should be initiated before the airplane reaches a safe altitude above the ground, at least 400 feet.

BRIEFING GUIDES

General

While the Learjet *AFM* does not specifically require before takeoff and approach briefings, such briefings are appropriate under some circumstances. The briefing guides presented below should be used when flying with unfamiliar crewmembers or any other time the PIC believes they are necessary.

It should be noted that many of these items may be briefed well before engine start. Many of them can be discussed before arriving at the airplane.

Pretakeoff Briefing

The pretakeoff briefing should address the following items:

- Type of takeoff: rolling or standing, flap setting, etc.
- Review takeoff data to include power setting and speeds.
- Procedures to be used in the event of an emergency before or after V_1 speed including emergency return procedures
- Heading and altitudes to be flown during the departure including restrictions, if any
- Radio, navigational systems, and flight director settings
- Anti-icing requirements, if applicable
- Specific PM duties and callouts. (See Takeoff Procedures, later in the chapter, for additional information.)
- A request for "Any questions?" directed to the other pilot

Approach Briefing

The approach briefing should be completed before starting descent and address the following items. The PF will normally transfer airplane control to the PM during the briefing.

- Approach to be used and backup approach, if available
- Special procedures to be used during the approach, such as circling approach procedures, interception of a radial from an arc VDP, etc.
- Altitudes of IAF, FAF, stepdowns, sector, and obstacles
- Minimums (DH, MDA), (HAT, HAA), radio altimeter setting
- Missed approach point and procedures timing to MAP/VDP
- Radio (COM/NAV) setup desired
- Anti-icing requirements
- Specific PM duties and callouts. (See Approach Procedures, later in this chapter, for additional information.)
- The procedure for transitioning to visual flight
- A request for "Any questions?" directed to the other pilot

At the completion of the Approach briefing, the PF announces "Approach briefing complete," and reassumes control of the airplane if control had been transferred to the PM.

TAKEOFF PROCEDURES

When cleared for takeof the PF calls for Runway Lineup checklist. The PM reports, "Runway Lineup checklist complete, cleared for takeoff." The PF advances power to the takeoff thrust detent. The PM confirms the N_1 setting matches the N_1 bug.

At V_1 speed, the PM calls, "V_1." The PF releases the thrust levers and puts both hands on the control column.

At V_R, the PM calls, "Rotate." The PF rotates airplane to a 9° nose-up pitch attitude.

With a positive rate of climb, the PM calls, "Positive rate," and the PF calls "Gear Up". The PM positions gear switch to up and calls "Gear selected up." The PM monitors the gear while it is retracting and reports, "Gear up," when retraction is complete.

Before V_{FE} (V_2 plus 20 knots minimum), the PF calls, "Flaps up, After Takeoff checklist." The PM positions the flap handle to up and calls, "Flaps selected up." The PM monitors the flaps while they are retracting and reports, "Flaps up," when retraction is complete. PM accomplishes the After Takeoff checklist, and reports "After Take-off checklist complete."

CLIMB AND CRUISE PROCEDURES

The PM will announce all assigned altitudes and set them in the Preselect altitude. The PM will also call out 1,000 feet above, or below, all assigned altitudes and altitude restrictions. These calls will normally be made by stating the existing altitude and the assigned altitude or restriction. For example, “Through 9,000 feet, cleared to 8,000” or “Through FL 460 for 470.” The PM will also announce other significant altitudes, such as, “Through 18,000 feet, altimeter 29.92.”

The PF will periodically announce his intentions and targets throughout the flight, such as “Accelerating to 250 knots,” “Turning right to 260° and descending to 3,000 feet.”

Any change in cockpit function will be announced by the pilot making the change and acknowledged by the other pilot. For example, the PM announces, “VOR number two set to Springfield and identified.” PF acknowledges, “VOR two on Springfield.” PF announces, “Autopilot engaged and coupled in altitude select and heading modes.” PM acknowledges, “Roger.”

Transfer of airplane control will be announced by the pilot initiating the change and acknowledged by the pilot assuming control. Specific target values will be provided to the pilot assuming control. For example, the PF announces, “Take the airplane for a minute. We’re climbing at 250 knots to 7,000 on a vector to the 045 radial.” PM acknowledges, “I’ve got the airplane, climbing at 250 knots to 7,000 on this heading until intercepting the 045 radial.”

APPROACH PLANNING

Approach planning and briefing should be accomplished during cruise. Review hazardous terrain, MEAs, and minimum sector altitudes. Complete and review performance data to include V_{REF} speed, landing distance, and approach climb speed.

The PF will direct the PM to obtain destination weather or obtain it himself. If the PM obtains the weather, the PF will normally assume ATC communications while the PM is obtaining weather. In either case, after checking weather, the pilot who did so will brief the other pilot on the destination weather, the expected approach, and any other significant information.

If a VDP has not been published, a “time to see the runway” may be computed as follows. Take the MDA, divided by 10, and subtract that, in seconds, from the time from the FAF to the MAP. For example, assume the MDA is 400 feet and the time from the FAF to the MAP is 1 minute and 45 seconds. Four hundred, divided by 10 equals 40. Subtracting that from 1:45 equals 1:05 from the FAF to see the runway. If the runway is not in sight at the end of that time, either a higher than normal rate of descent will be required, or the airplane will land beyond the normal touchdown zone.

Normally, ATC will determine when a descent may be started. However, descents may sometimes be started at the PF’s discretion. To determine how far out to start descent for an approach, use 3 times the altitude to be lost, divided by 1,000. For example, to lose 40,000 feet, 3 times 40,000 equals 120,000, divided by 1,000 equals 120 miles out to start descent.

The Descent checklist should be started before, or early in, the descent.

DESCENT PROCEDURES

The same procedures used during climb and cruise will be used during descent. The PM will accomplish the Descent checklist, as directed by the PF, and make appropriate altitude callouts.

APPROACH PROCEDURES

The PF should initiate the Approach checklist when descending out of 18,000 feet or when within 50 miles of the destination airport. The checklist will be accomplished so as to not interfere with the visual lookout for other traffic.

Configuration changes during the approach will be accomplished using the same crew coordination techniques used after takeoff. The PF calls for a configuration change. The PM acknowledges, selects the switch position, monitors and reports when gear and flaps are in the selected positions.

The Approach checklist will be completed and the airplane slowed to V_{REF} + 30 knots (minimum) before reaching the IAF.

Over the IAF, for other than a straight-in approach, the PF will turn outbound, call for flaps 8°, slow the airplane to V_{REF} + 25 knots (minimum) and begin a descent, if necessary.

Wings-level outbound in the procedure turn, the PM will start timing, announce the time to be flown and the next heading and altitude. At the expiration of the procedure turn outbound time, the PM will announce the time is up, the direction of turn, the next heading and altitude, if an altitude change is required. For example, “Time’s up, right turn now to 225° and cleared down to 3,000.”

Approaching the final approach course, the PM will monitor the CDI or bearing pointer and report “CDI alive,” or within 5° of the inbound course.

Established on final approach (within 2 1/2° for all approaches), the PF will call for flaps 20°, slow the airplane to V_{REF} + 20 knots (minimum), and begin a descent, if necessary. After the flaps have been set to 20°, the PF will call, “Gear down, Before Landing checklist.” The PM extends the landing gear, completes the Before Landing checklist up to flaps down and reports, “Before Landing checklist complete to flaps down.”

Over the FAF, for straight in on a two-engine approach, the PF will call for flaps down, slow the airplane V_{REF} (minimum), and begin a descent. (For a single-engine, the flaps remain at 20°.) The PM will begin timing, extend the flaps, and complete the Before Landing checklist. The PM will also confirm that the COM/NAV radios are set properly, check the flight instruments, airspeed bugs, preselect altitude display, MDA and/or DH displays on both PFDs. He will then report, “Before Landing checklist complete, no flags, cleared for descent to _______ feet.”

After passing the FAF, the PM will begin looking for visual references outside the airplane. However, he will also monitor the instruments and call out significant deviations such as one dot or more, deflection on the CDI or glideslope, and airspeed variations greater than –0 to +10 knots for V_{REF}. If the PF does not respond to the call out, the PM will repeat it. If the PF does not respond to the second call out, the PM should assume the PF has been incapacitated and announce that he (the PM) is taking control of the airplane.

The PM will call out the time to the VDP/MAP and 1,000, 500, and 100 feet above MDA or DH. The PM will also report visual contact with the ground such as, “Visual contact, no runway yet,” “Approach lights in sight at 11 o-clock,” or “Runway in sight straight ahead.”

Approaching minimums, or the missed approach point, the PF will begin cross-checking outside the airplane for visual references. When satisfied that visual references are adequate for landing, the PF will announce, “I’m going visual,” or “Going outside.” At this point, the PM directs his attention primarily inside the airplane, while cross-checking outside, and calls airspeed, descent rate, and altitude. The purpose is to provide the PF, verbally, the same information he would have if he were still on instruments.

Airspeed should be called as plus or minus V_{REF}, descent rate as up or down, and altitude above the ground. For example, “Plus 5, down 500, 100 feet,” indicates the airspeed is V_{REF} plus 5 knots, the airplane is descending at 500 feet per minute, and is 100 feet above the ground.

GO-AROUND/BALKED LANDING

If a go-around/balked landing is necessary, the PF will call "Going around, flaps 8°" and simultaneously disengage the autopilot, select flight director go-around mode, establish a 9° nose-up pitch attitude, set power to takeoff thrust, or as required, and check the spoilers are retracted. The PM will set, or confirm, the flaps at 8°, call out the direction of turn, if one is required, and the missed approach heading and altitude. The PM notifies ATC of the missed approach.

MANEUVERS

GENERAL

This section contains a description of most of the maneuvers that are likely to be encountered during Learjet training and operational flying. While there is always more than one way to fly an airplane, these procedures have been developed over many years of Learjet operations. They have proven to be safe, efficient, and readily manageable. These procedures are consistent with the *AFM*. However, if a conflict should develop between these procedures and those in the *AFM*, the *AFM* procedures should be used.

PERFORMANCE STANDARDS

The performance standards in Table 18-1 should be maintained during all Learjet flight operations.

MINIMUM MANEUVERING SPEEDS

Minimum maneuvering speeds are expressed in terms of V_{REF} speed which is 1.3 times the stalling speed in the landing configuration.

For maneuvering with up to 30° of bank, the following minimum speeds should be used:

Spoilers deployed:	V_{REF} + 40 knots
Flaps UP:	V_{REF} + 30 knots
Flaps 8°:	V_{REF} + 25 knots
Flaps 20°:	V_{REF} + 20 knots
Flaps DN:	V_{REF} + 10 knots

For maneuvering with up to 15° of bank, on final approach for landing for example, the following minimum speeds should be used:

Spoilers deployed:	V_{REF} + 30 knots
Flaps UP:	V_{REF} + 20 knots
Flaps 8°:	V_{REF} + 15 knots
Flaps 20°:	V_{REF} + 10 knots
Flaps DN:	V_{REF}

THRUST SETTINGS

Actual thrust settings will vary depending upon the temperature, pressure altitude, and airplane gross weight. The following target settings are approximate, but may be used to provide a starting point to determine the actual power setting.

Below 10,000 MSL, 60% N_1 to maintain 200 KIAS, 72% N_1 to maintain 250 KIAS.

Between 10,000 MSL and FL 250, 72 to 75% N_1 to maintain 250 KIAS.

Table 18-1. PERFORMANCE STANDARDS

Takeoff:	
Normal/Crosswind	Heading ± 5°
	Airspeed ± 5 Knots of appropriate V-Speed, assigned or airspeed restriction.
Instrument	Same as above
Single-Engine	Same as above
Departure and Holding:	
Airspeed	± 10 Knots
Heading	± 10°
Altitude	± 100 Feet
Accurately tracks course, radial or bearing	
Steep Turns:	
45° Bank (min)	± 5°
Altitude	± 100 Feet
Airspeed	± 10 Knots
Heading at rollout	± 10°
Approaches to Stalls:	
One turning using 15–30° of bank	
Powerplant Failure:	
Altitude	± 100 Feet
Airspeed	± 10 Knots
Heading	± 10°
All Approaches:	
Prior to final maintains:	
Altitude	± 100 Feet
Airspeed	± 10 Knots
Heading	± 5°
Accurately tracks radials, courses and bearings	
Precision final approach:	No more than 1/4 scale deflection (1/2 Dot) of either glideslope or localizer
Airspeed	± 5 Knots
Non-Precision Approaches:	
No more than 1/4 scale deflection (1/2 Dot) of course deviation indicator (CDI) or + 5° in case of RMI/bearing pointer	
Airspeed	± 5 Knots
MDA	Maintains –0 to +50 Feet to MAP
Circling Approach:	
Airspeed/V-Speed	± 5 Knots
Heading/Track	± 5°
Altitude	Maintains –0 to +100 Feet
Missed Approach:	
Altitude	± 100 Feet
Airspeed	± 5 Knots
Heading	± 5°
Accurately tracks courses, radials and bearings	
Landing:	
Touchdown	500–3,000 Feet past runway threshold, not to exceed 1/3 of runway length, centerline between gear

TAKEOFF

Either 8 or 20° of flaps may be used for takeoff. The normal, standing takeoff (Figure 18-1) must be used to achieve the performance specified in the *AFM*. If the runway available is at least 10% longer than the planned takeoff distance, a rolling takeoff (Figure 18-2) may be used. The procedures are the same except, for a standing takeoff, thrust is set before brake release. For a rolling takeoff, the brakes are released before the thrust is set. During a rolling takeoff, takeoff thrust must be set before the runway remaining equals the takeoff distance.

The flaps are retracted before V_{FE} (V_2 plus 20 knots minimum) and then the After Takeoff checklist is normally accomplished. However, if traffic conditions warrant, the After Takeoff checklist may be delayed until the airplane is clear of local traffic.

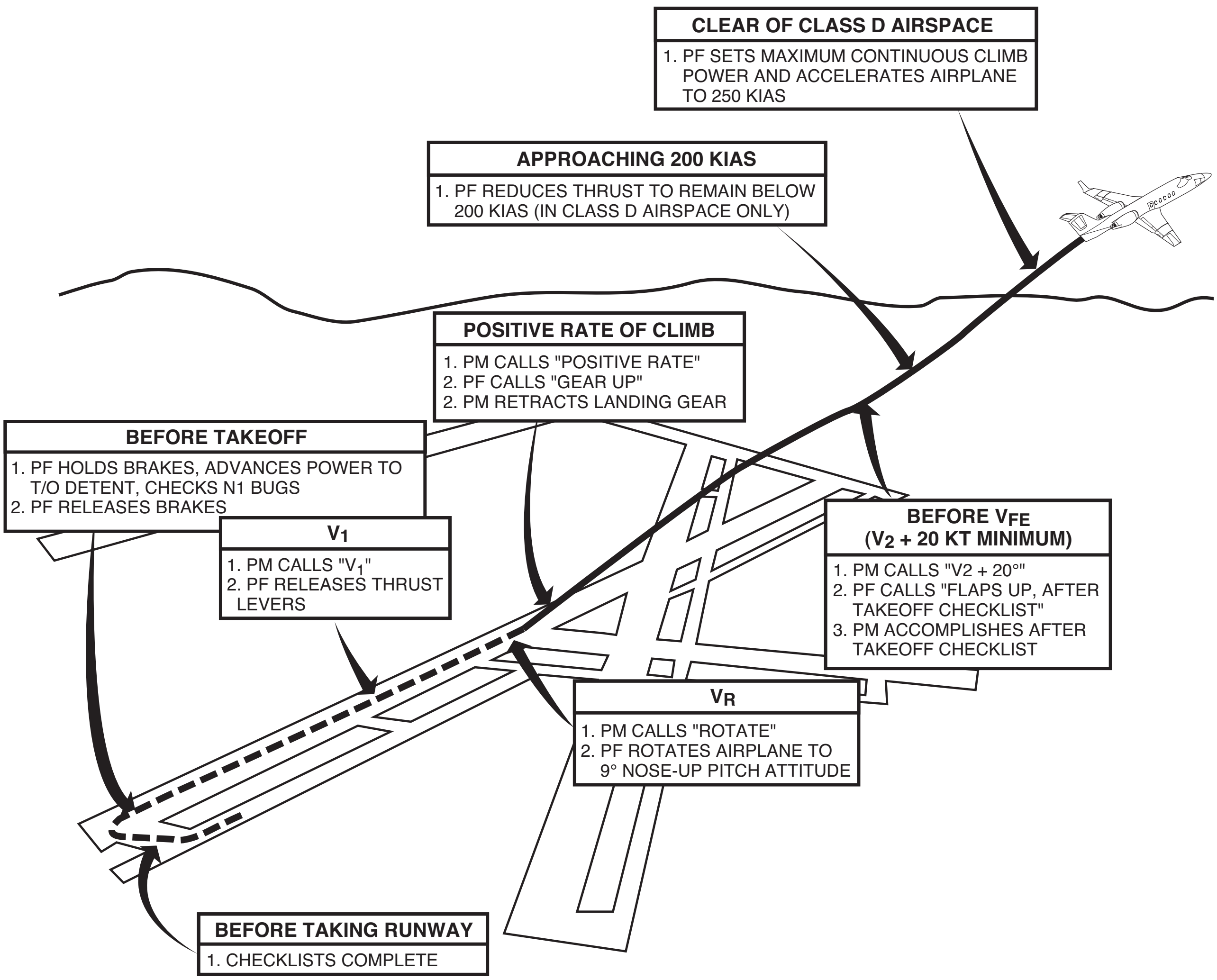


Figure 18-1. Standing Takeoff

Approaching 200 knots, the PF should adjust thrust and pitch attitude if necessary to maintain 200 knots or less within the ATA (Class D airspace). For passenger comfort and ease of airplane control, it is recommended that the pitch attitude not exceed 20° nose-up.

The maximum continuous climb thrust setting is a variable depending on temperature and pressure altitude. The MCT detent should be used. The Maximum Continuous Thrust (N_1) chart, in the Performance Data section of the checklist, and the *AFM* may be cross-checked.

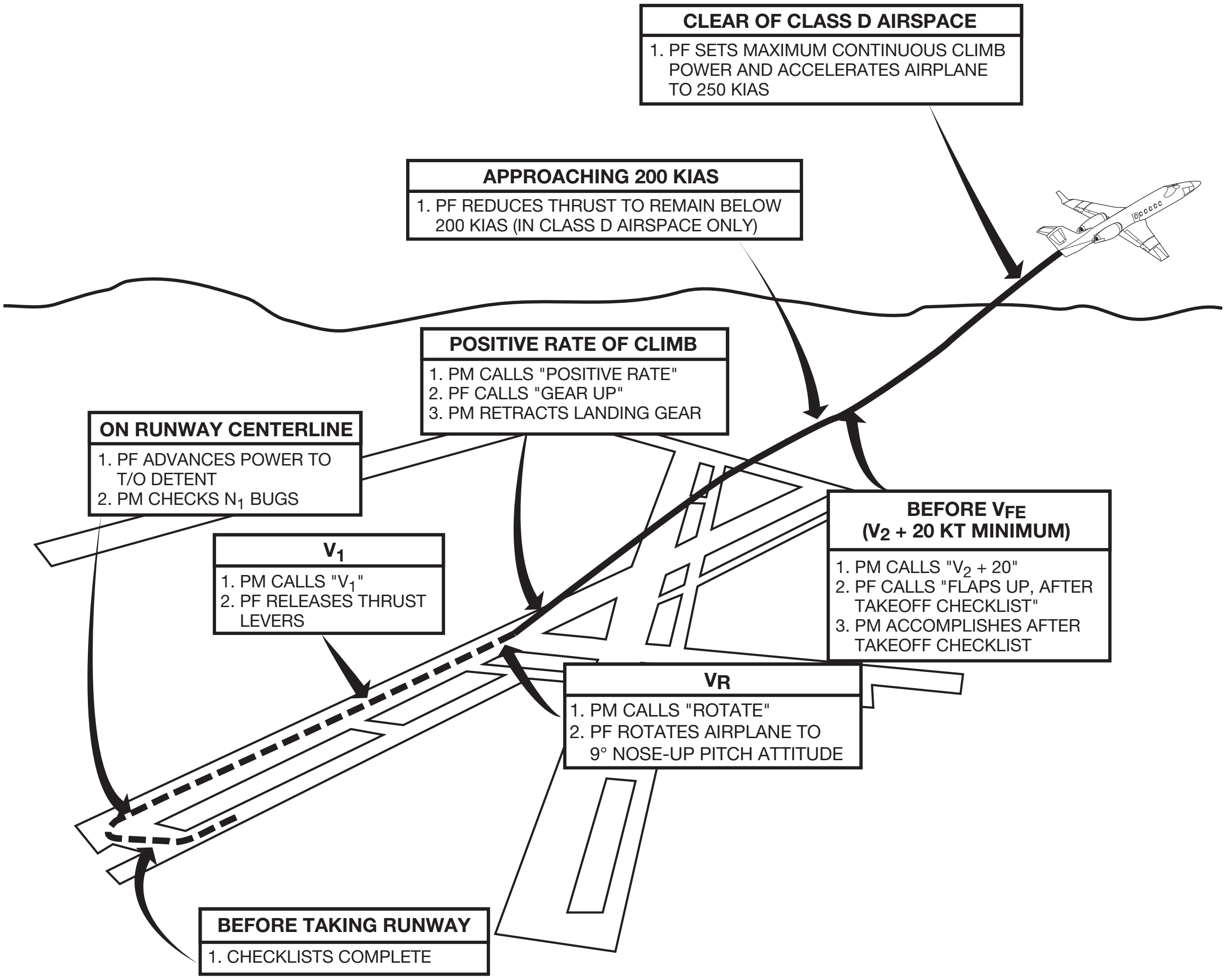


Figure 18-2. Rolling Takeoff

ENGINE FAILURE BELOW V_1 SPEED

If an engine fails below V_1 speed (Figure 18-3), the takeoff must be aborted. The PF will simultaneously apply maximum braking, reduce thrust levers to idle, and extend the spoilers. The thrust reversers may be deployed if necessary.

Takeoffs may be aborted for malfunctions other than engine failure; however, the same procedure should normally be used.

ENGINE FAILURE ABOVE V_1 SPEED

If an engine fails above V_1 speed (Figure 18-4), the takeoff will normally be continued. The PF will maintain directional control with ailerons and rudder and keep the nosewheel on the runway until reaching rotate speed. After liftoff, the initial climb will be made at V_2 speed with takeoff flaps until the airplane is clear of obstacles or, if there are no obstacles, to 1,500 feet AGL. The PF will then accelerate the airplane to V_2 plus 20 knots (mini-

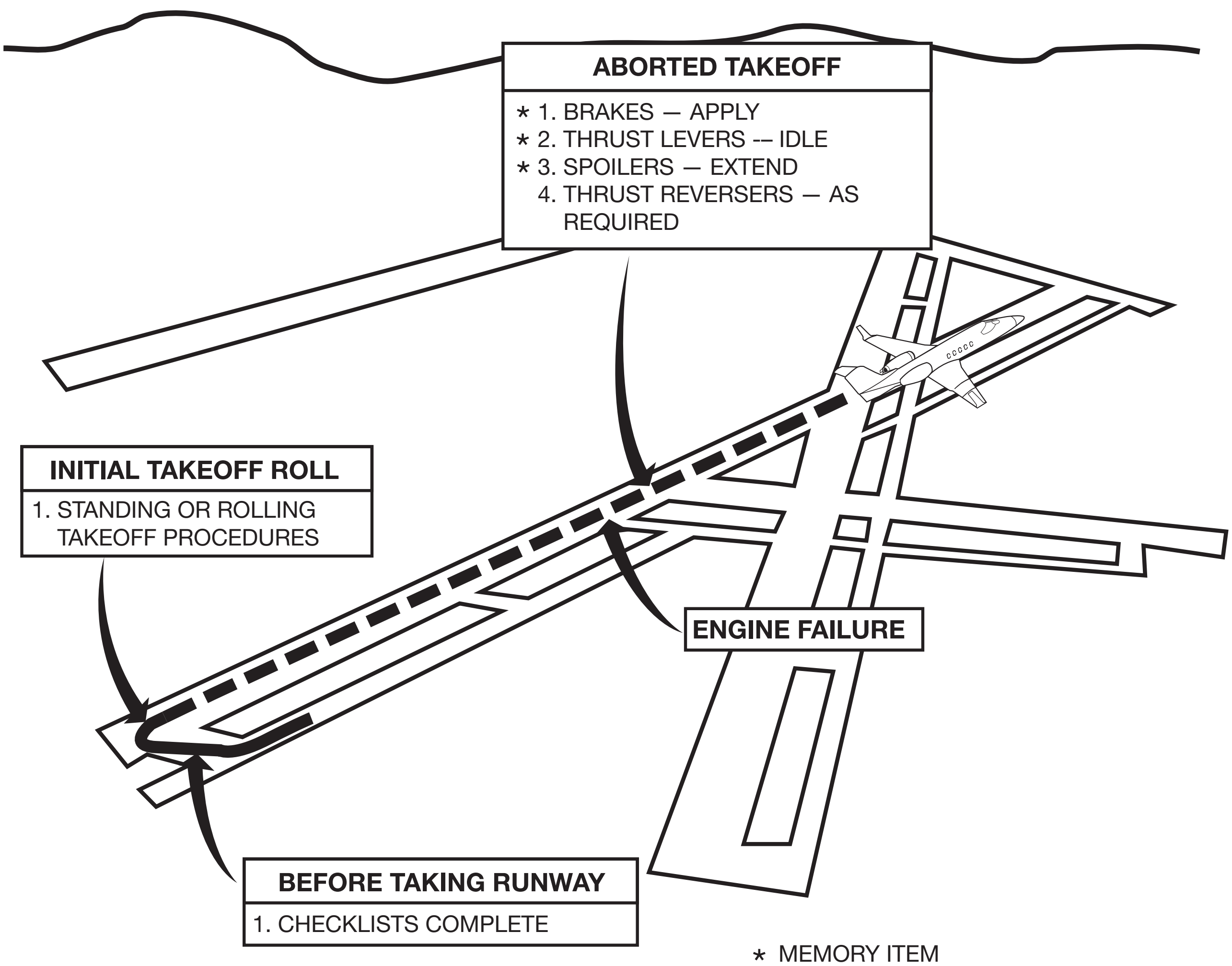


Figure 18-3. Engine Failure Below V_1 Speed

mum) and direct the PM to retract the flaps. The PF will then accelerate the airplane to single-engine climb speed (normally 200 knots) and climb to assigned altitude.

At a safe altitude above the ground (no lower than 400 feet), the memory items for the appropriate In Flight checklists will be completed. The rest of the appropriate In Flight checklists and the After Takeoff checklist will normally be completed at or above 1,500 feet AGL. The crew may then elect to obtain clearance to return to the departure airport for landing or to proceed to an alternate airport.

STEEP TURNS

Steep turns (Figure 18-5) are used to build confidence in the airplane and improve instrument cross-check. They may be accomplished at any altitude above 5,000 feet AGL. The higher the altitude, the more difficult the maneuver is to perform correctly. Steep turns are accomplished without flight director steering commands since the flight director will not command 45° of bank.

Thrust must be increased approximately 2% N_1 to maintain airspeed 250 knots during steep

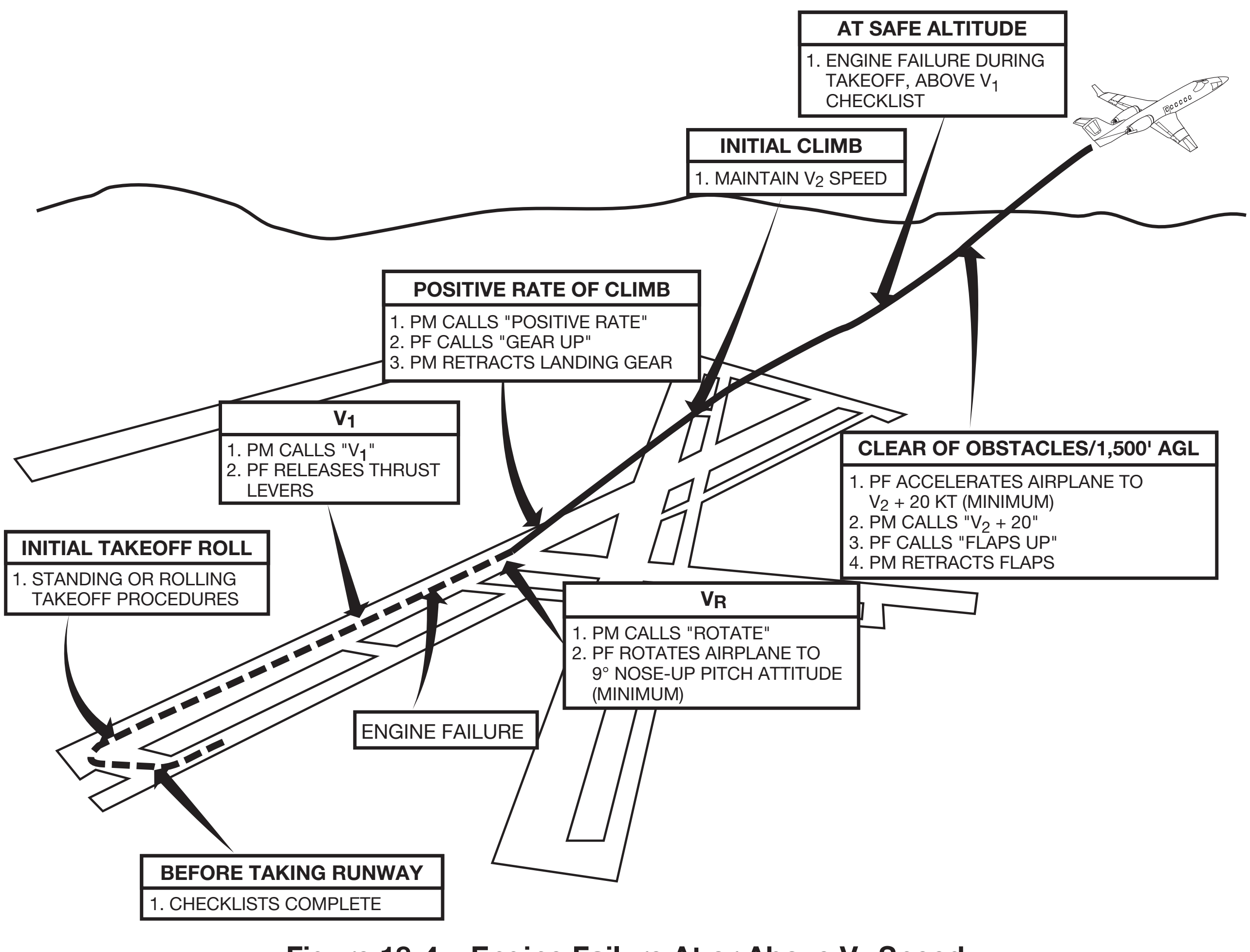


Figure 18-4. Engine Failure At or Above V_1 Speed

18 MANEUVERS AND PROCEDURES

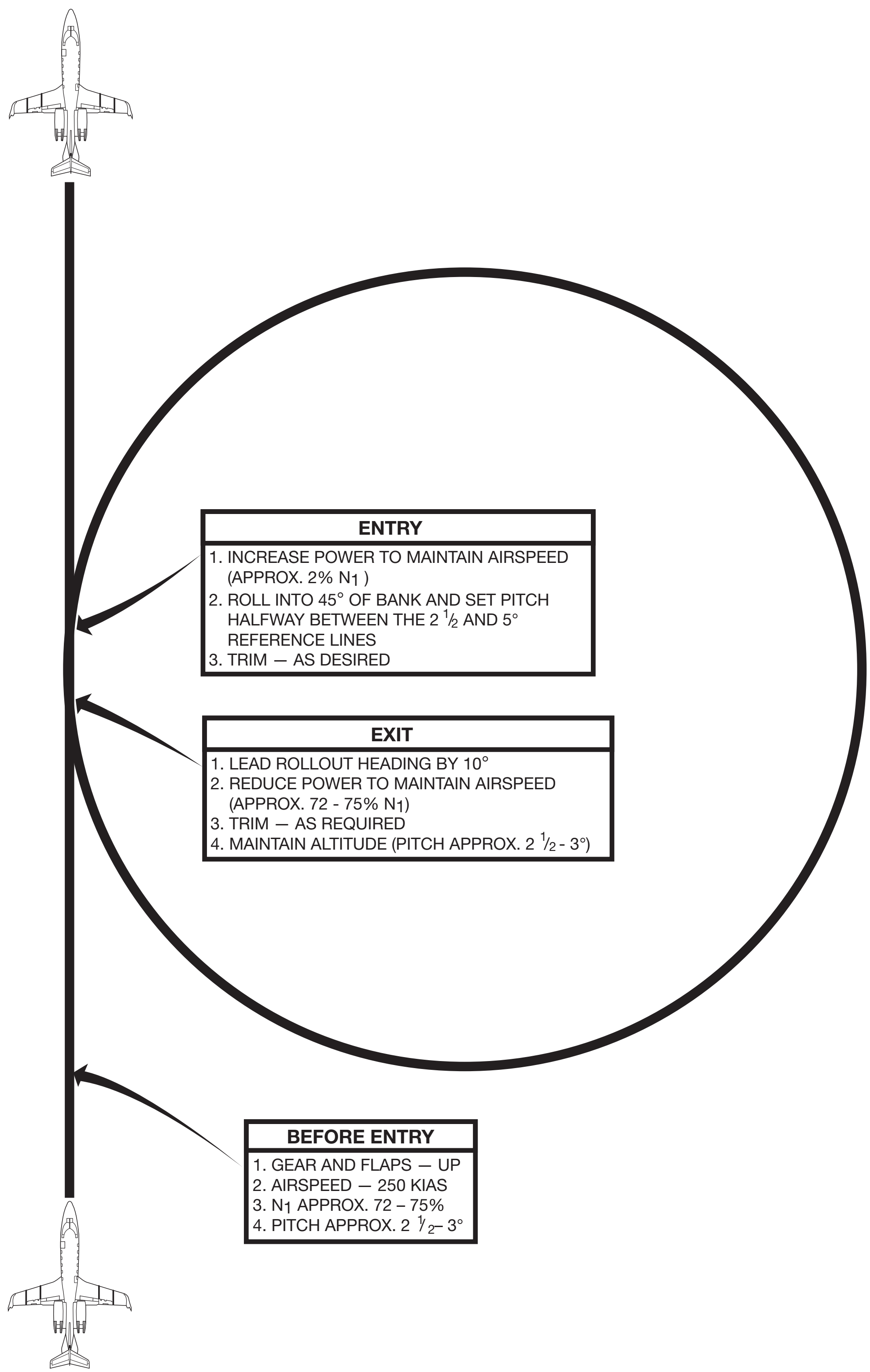


Figure 18-5. Steep Turns

turns. The airplane should be kept in trim and the bank angle should be held constant. If altitude corrections are necessary, they should be made in pitch only. It is not necessary to shallow the bank to climb during a steep turn in a Learjet.

Steep turns of at least 180°, preferably 360°, should be practiced in each direction.

UNUSUAL ATTITUDE RECOVERY, NOSE-HIGH, LOW-SPEED

Recovery from a nose-high, low-speed unusual attitude (Figure 18-6) should be made while maintaining positive g forces and without stalling the airplane. It is accomplished by increasing thrust while simultaneously increasing the angle of bank, not to exceed 90°, to allow the nose of the airplane to descend to

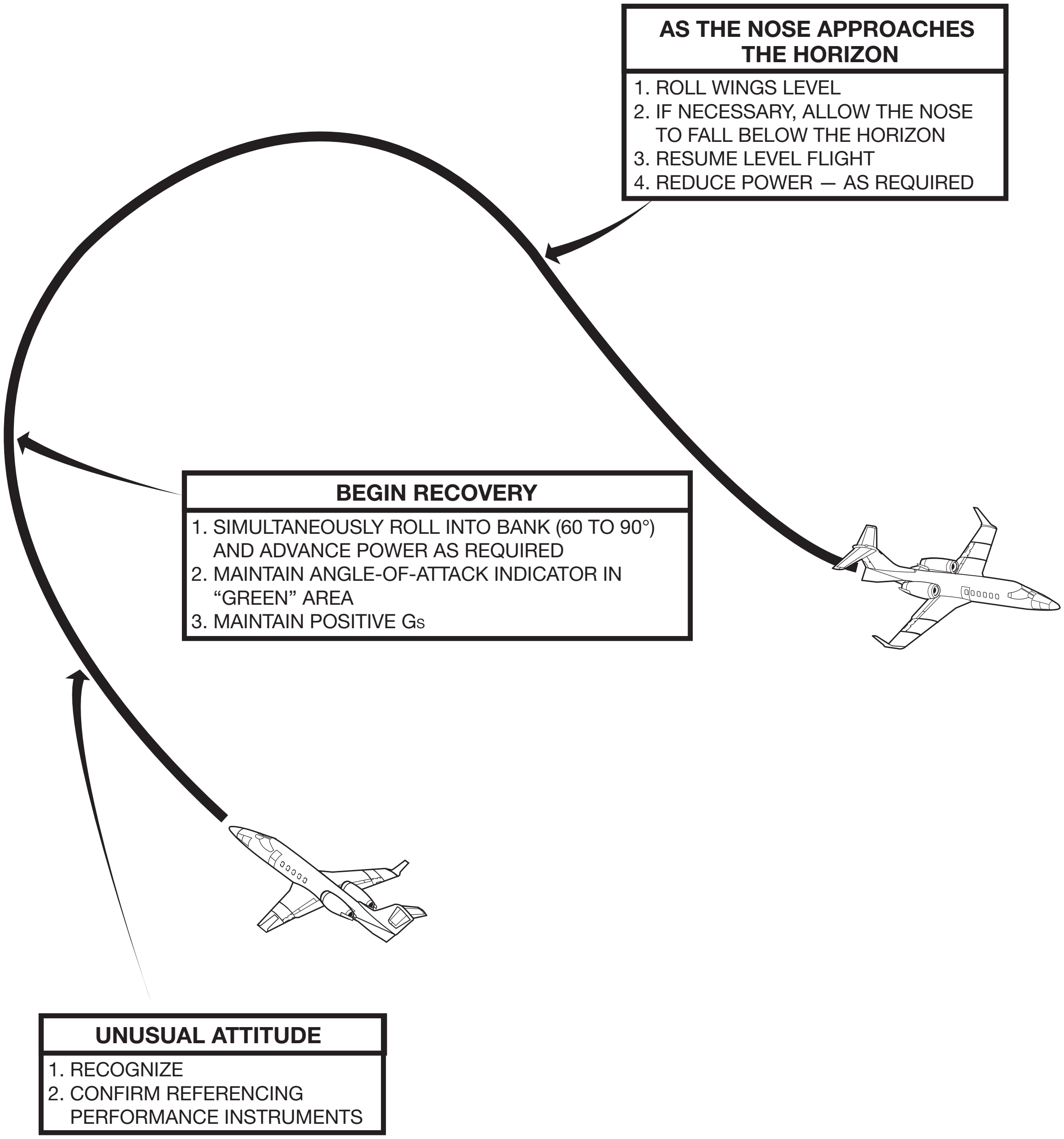


Figure 18-6. Unusual Attitude Recovery—Nose-High, Low Speed

the horizon without negative g forces. The attitude indicator should be used during the recovery and the angle-of-attack indicator cross-checked to maintain the pointer in the green band.

UNUSUAL ATTITUDE RECOVERY, NOSE-LOW, HIGH-SPEED

Recovery from a nose-low, high-speed unusual attitude (Figure 18-7) should be made with minimum loss of altitude while keeping the airspeed below V_{MO} or M_{MO}. It is accomplished by simultaneously reducing thrust to idle and rolling the wings level. When the bank is less than 90°, elevator and pitch trim (if required) are used to raise the nose to the horizon. Spoilers should not be used during recovery from a nose-low unusual attitude.

During training, nose-low, high-speed unusual attitudes will always be presented so the airplane may be recovered without exceeding any limitations. However, during recovery from an actual, inadvertent, nose-low, high-speed unusual attitude, an overspeed condition may develop. In this case, the overspeed recovery procedures in the *AFM* should be used.

SLOW FLIGHT

Slow flight is used to develop the pilot's sense of feel for the airplane's low-speed handling characteristics and improve the pilot's coordination and instrument cross-check. Slow flight is accomplished in the clean, takeoff, and landing configurations (Figures 18-8, 18-9, and 18-10), and is normally accomplished between 10,000 and 15,000 feet MSL. Slow flight should not be accomplished below 10,000 AGL.

Slow flight may be practiced while maintaining a constant altitude and heading or while maintaining a constant altitude and making turns to preselected headings. Slow flight may also be practiced while making constant rate climbs and descents to preselected altitudes. Slow flight practice may be terminated by a recovery to normal cruise or an approach to stall.

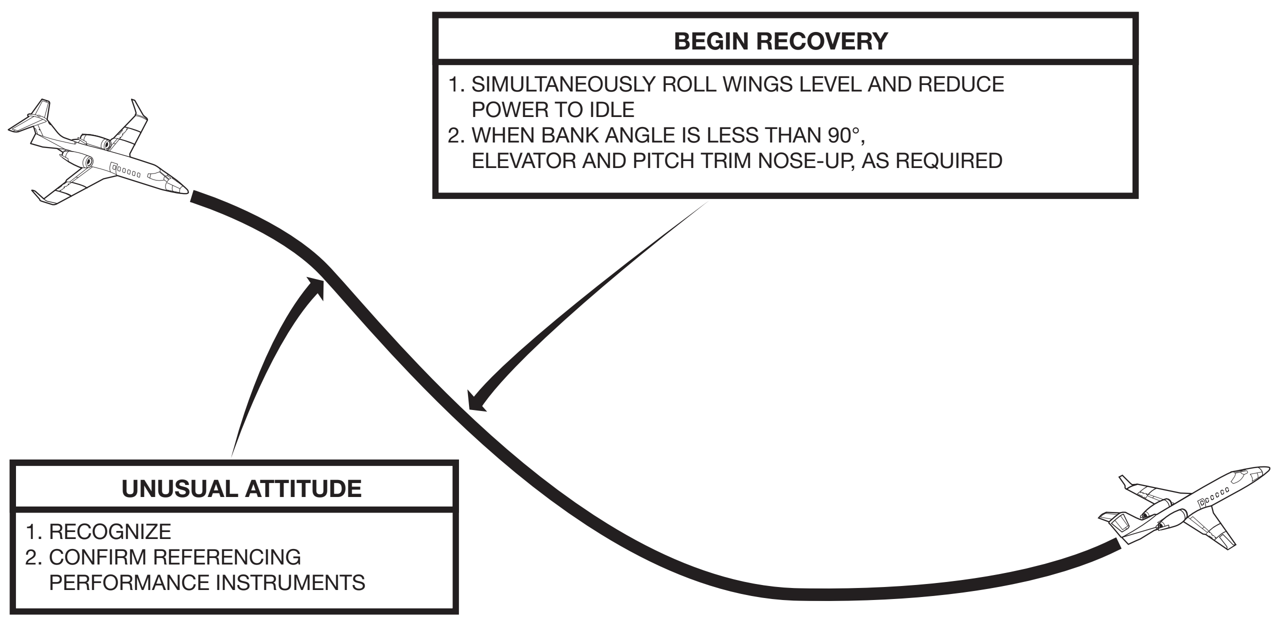


Figure 18-7. Unusual Attitude Recovery—Nose-Low, High Speed

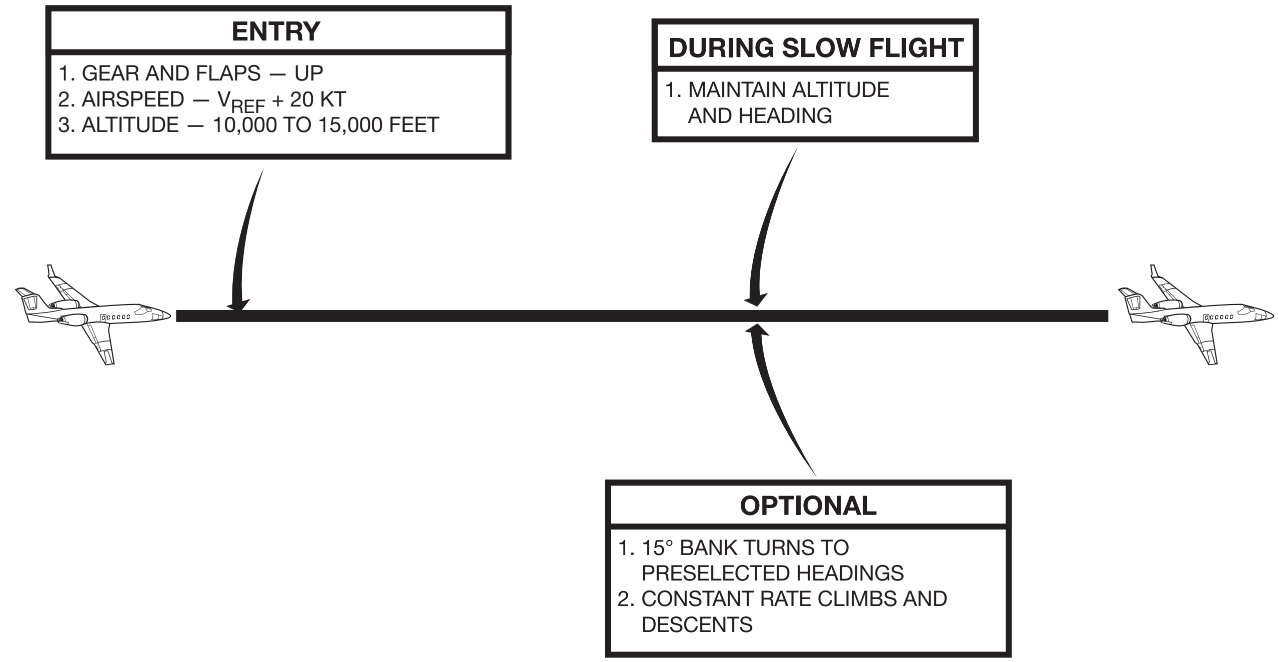


Figure 18-8. Slow Flight—Clean Configuration

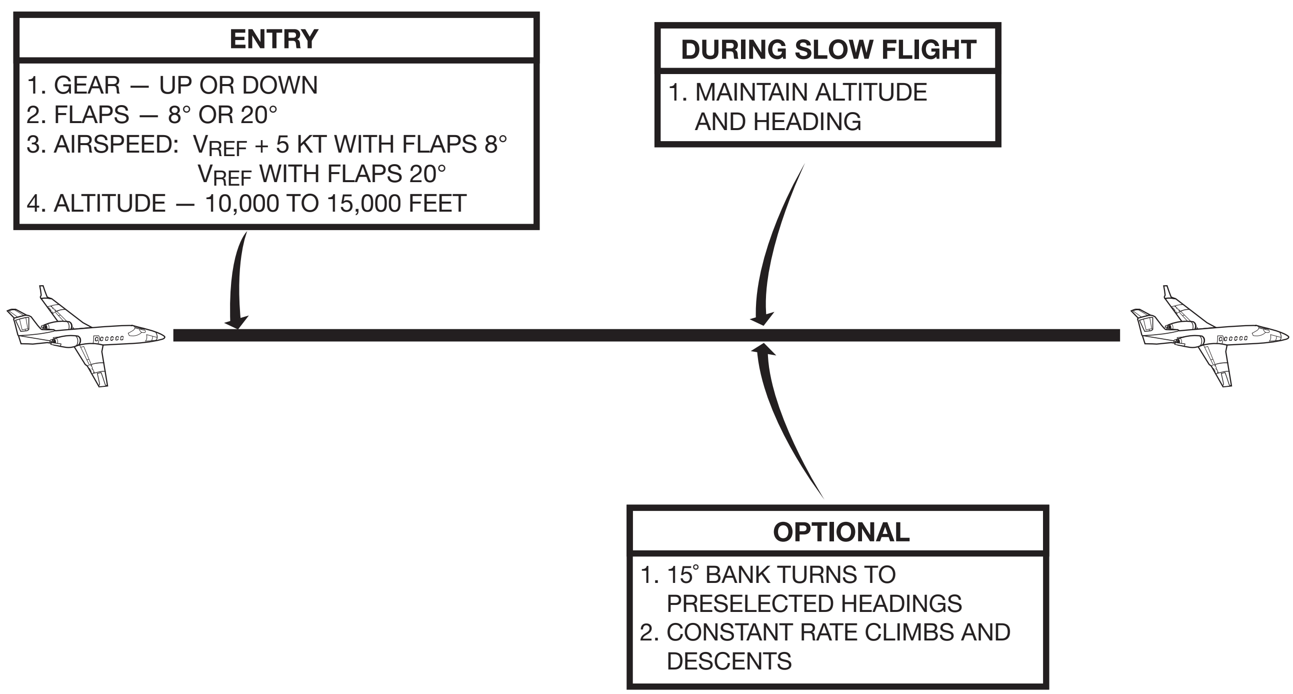


Figure 18-9. Slow Flight—Takeoff Configuration

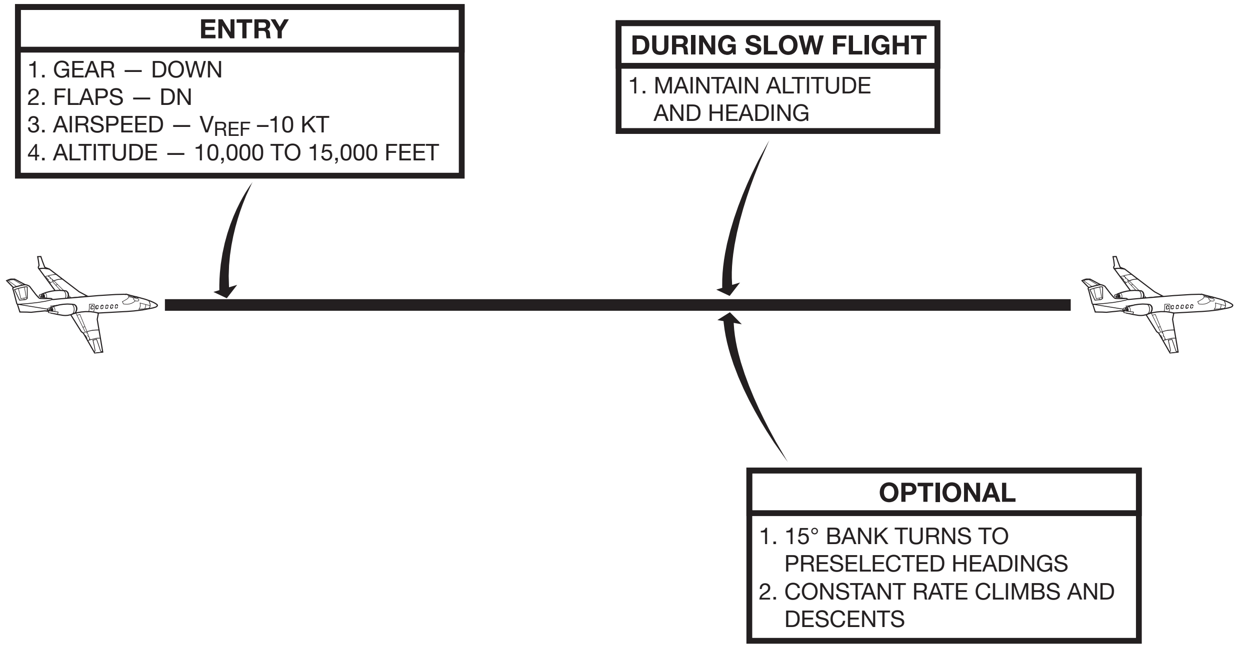


Figure 18-10. Slow Flight—Landing Configuration

APPROACH TO STALL

Approaches to stalls are accomplished in the clean, takeoff, and landing configurations (Figures 18-11, 18-12, and 18-13), and are normally accomplished between 10,000 and 15,000 feet MSL. Approaches to stalls should not be accomplished below 10,000 feet AGL. Approaches to stalls may be made from level or turning flight with 15 to 30° of bank. Approaches to stalls may also be combined with slow flight practice. All recoveries are made with power and minimum loss of altitude.

If the aircraft is allowed to decelerate below stall warning angle of attack, priority must be given to immediately reducing the angle of attack rather than attempting to maintain altitude. Stall recovery from typical cruise altitudes and other relatively low thrust conditions will require a significant loss of altitude.

Approach to stall recovery is initiated at the first indication of an impending stall. This indication is provided by the stick shaker and stall warning annunciator lights which activate as the angle-of-attack indicator needle moves into the yellow band.

Thrust levers should be advanced to the takeoff detent and the existing pitch attitude should be lowered to reduce the angle of attack. The wings should be leveled and altitude loss minimized as the aircraft accelerates out of the stall condition.

Approaches to stall from the landing configuration are normally terminated by a simulated missed approach (Figure 18-13).

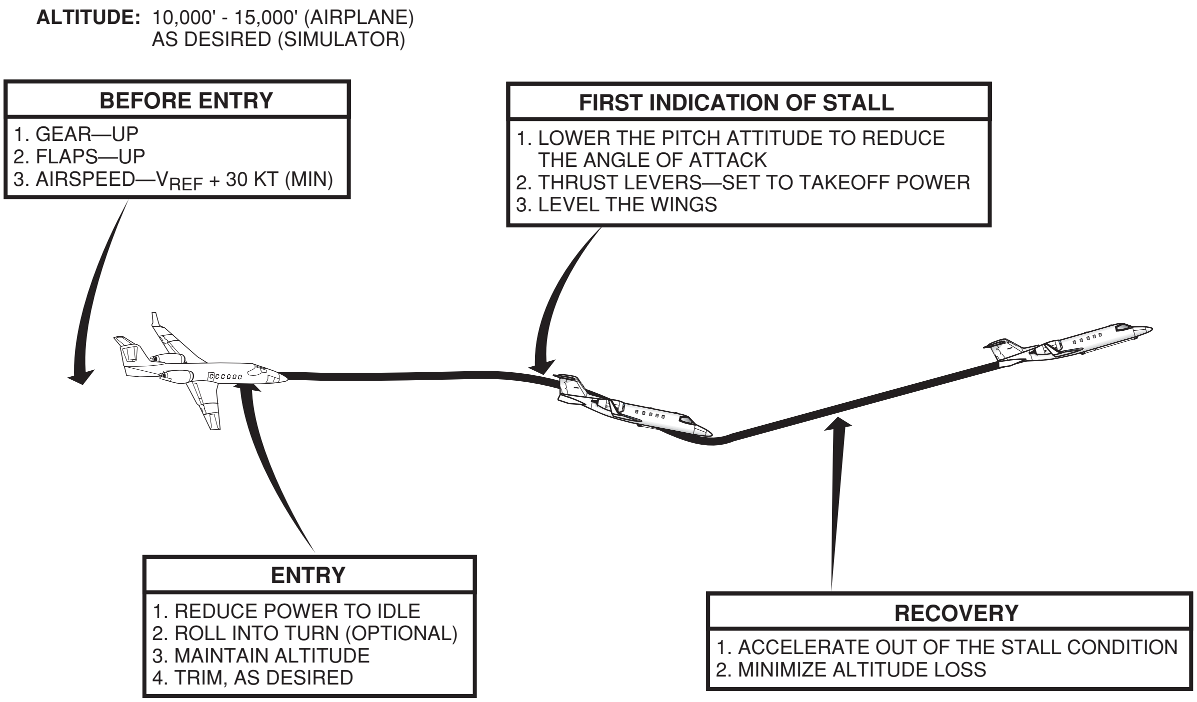


Figure 18-11. Approach to Stall—Clean Configuration

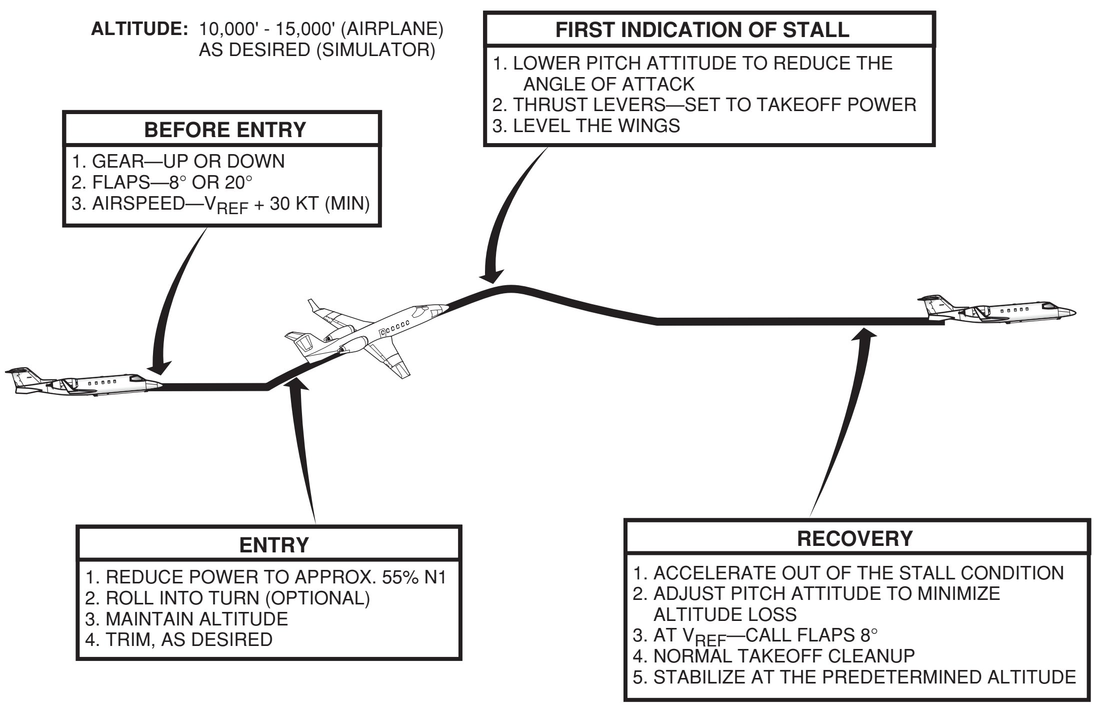


Figure 18-12. Approach to Stall—Takeoff Configuration

EMERGENCY DESCENT

Emergency descents are accomplished in accordance with *AFM* procedures as shown in Figure 18-14. The PF should accomplish the checklist memory items and allow the airplane to pitch down to a 10 to 15° nose-down pitch attitude. This pitch attitude is maintained until the airplane accelerates to M_{MO}/V_{MO}. Then the pitch attitude is adjusted to maintain M_{MO}/V_{MO}.

After the emergency descent has been established, the crew should determine the desired level off altitude.

NOTES

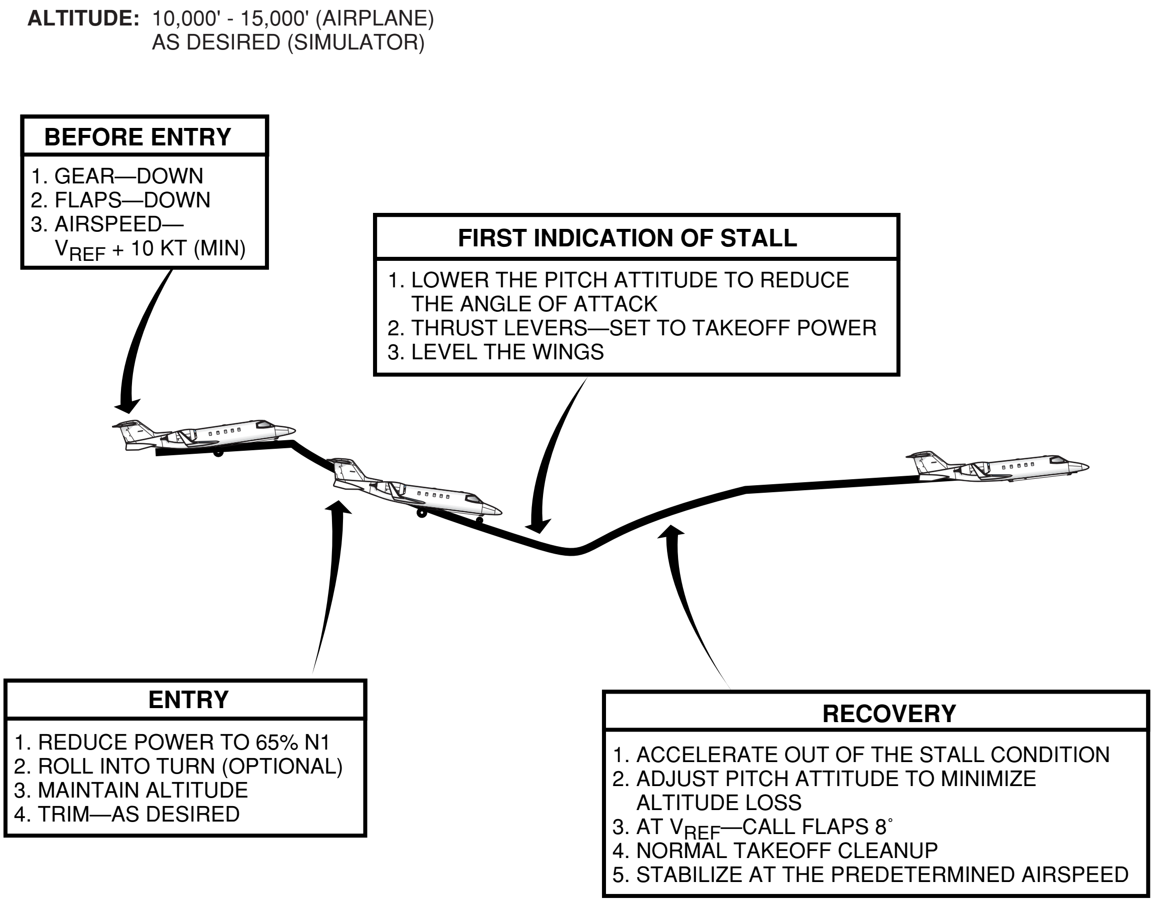


Figure 18-13. Approach to Stall—Landing Configuration

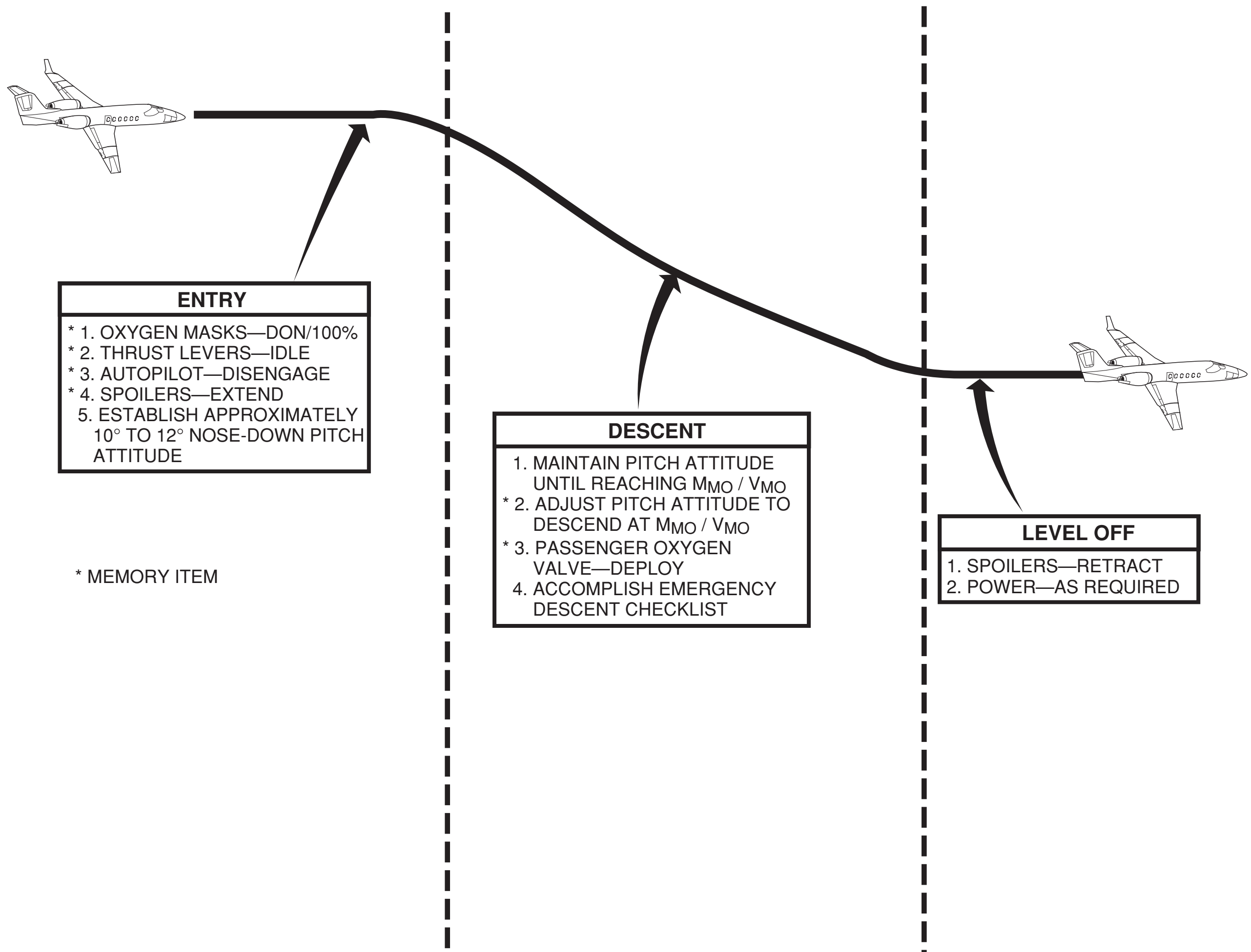


Figure 18-14. Emergency Descent

VISUAL TRAFFIC PATTERN, TWO ENGINES

A two-engine visual traffic pattern is shown in Figure 18-15. The airspeeds indicated on the diagram are minimums. Traffic pattern altitude for jet airplanes is normally 1,500 feet AGL. During gusty wind conditions, 1/2 the gust velocity should be added to V_{REF} on final approach. If a crosswind exists, final approach should be flown with a drift correction angle (crab) to maintain alignment with the runway centerline. Approaching touchdown, rudder should be applied to align the airplane with the runway centerline and the upwind wing lowered with aileron to prevent drift.

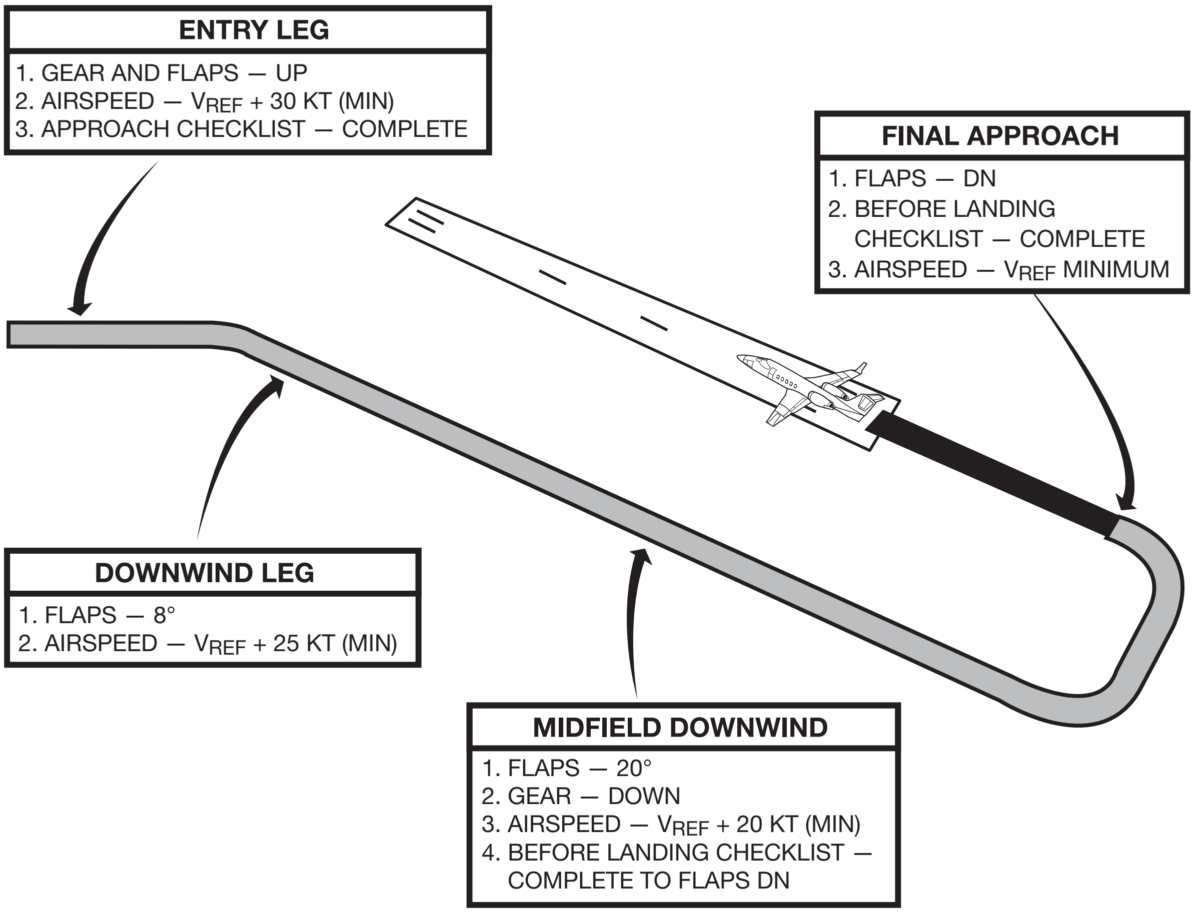


Figure 18-15. Visual Traffic Pattern—Two Engines

VISUAL TRAFFIC PATTERN, SINGLE ENGINE

A single-engine visual traffic pattern (Figure 18-16) is flown exactly the same as a two-engine pattern except for the flap setting on final approach. For a single-engine landing, final approach is flown with flaps 20° at abnormal landing speed for the flap setting. Additionally, the PF may elect to have the PM remove some, or all, of the rudder trim on final approach.

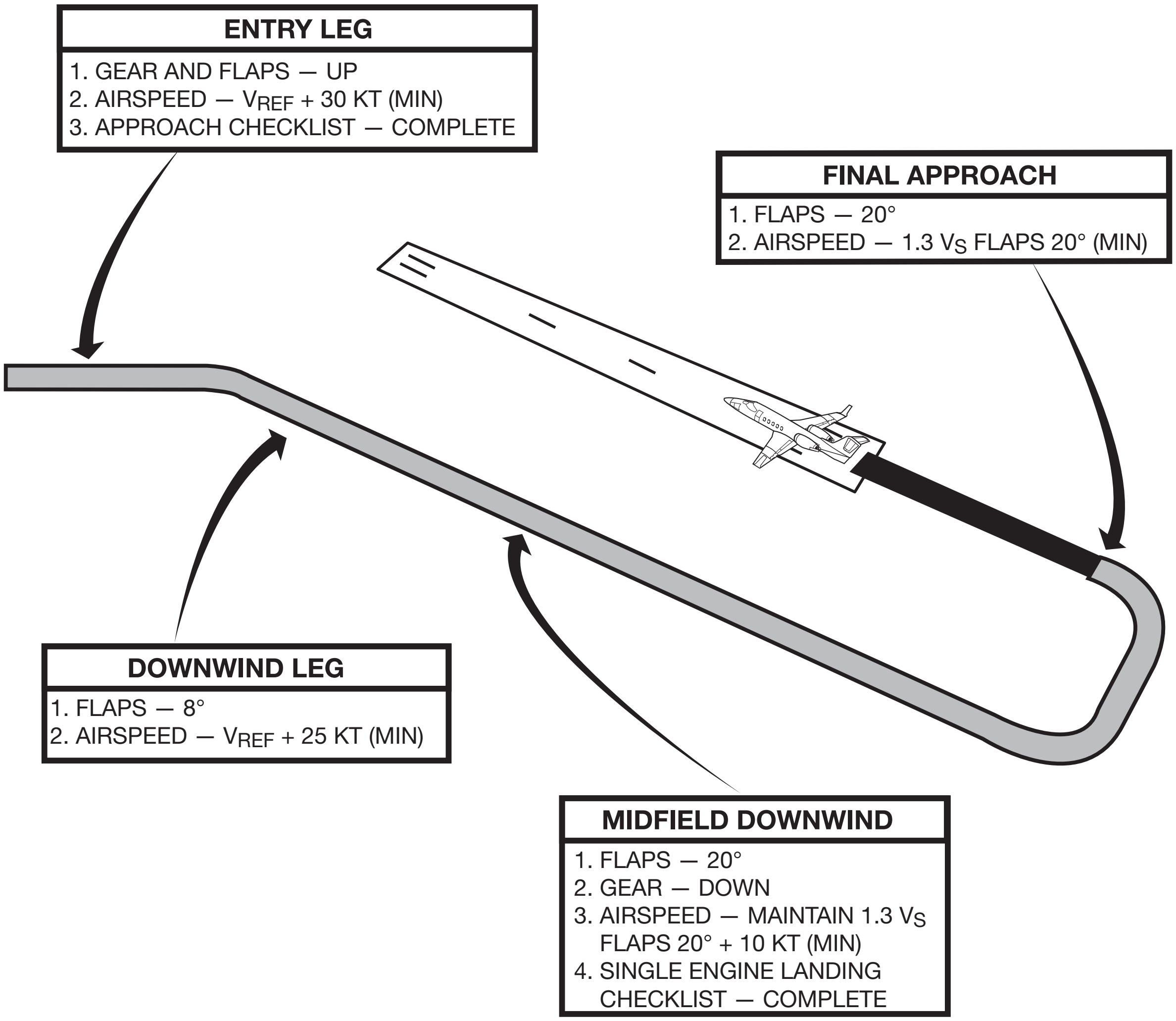


Figure 18-16. Visual Traffic Pattern—Single Engine

FLAPS UP LANDING

The correct landing distance for a flaps up landing (Figure 18-17) is determined by multiplying the normal landing distance by 1.2. Consideration should be given to reducing the airplane's weight, if possible, to lower the landing speed and reduce landing distance if the available runway length is marginal.

Final approach is flown at abnormal landing speed for flaps up. To avoid excessive floating during the landing flare, the PF should establish the landing attitude as thrust is reduced to idle, maintain the attitude, and allow the airplane to touch down. The use of the thrust reversers is recommended during a flaps up landing.

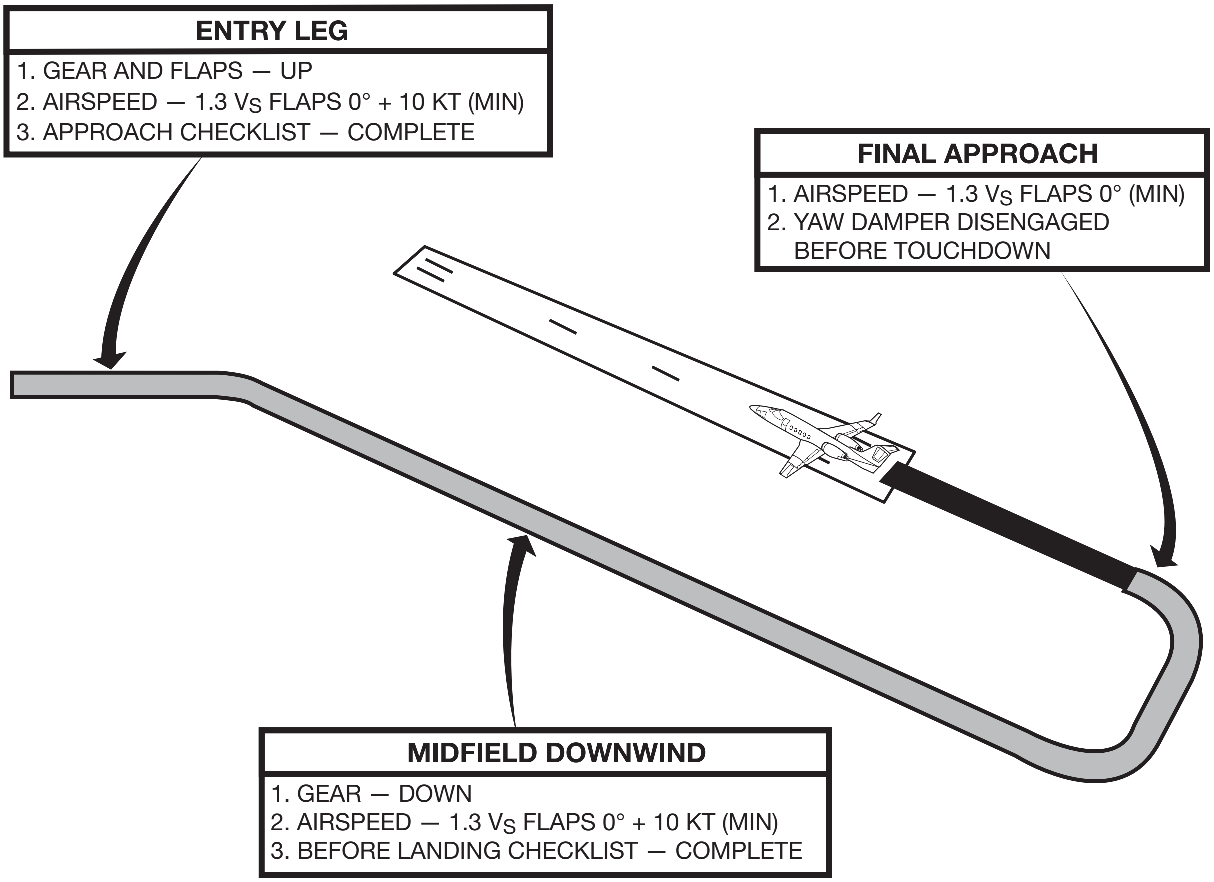


Figure 18-17. Flaps Up Landing

PRECISION INSTRUMENT APPROACH

A typical precision instrument approach is shown in Figure 18-18. All accepted instrument flying procedures and techniques should be used while making instrument approaches in the Learjet.

Two-engine, precision approaches should be flown with a stabilized airspeed and configuration from the FAF inbound. Single-engine, precision approaches should be flown at flaps 20° abnormal landing speed from the FAF inbound.

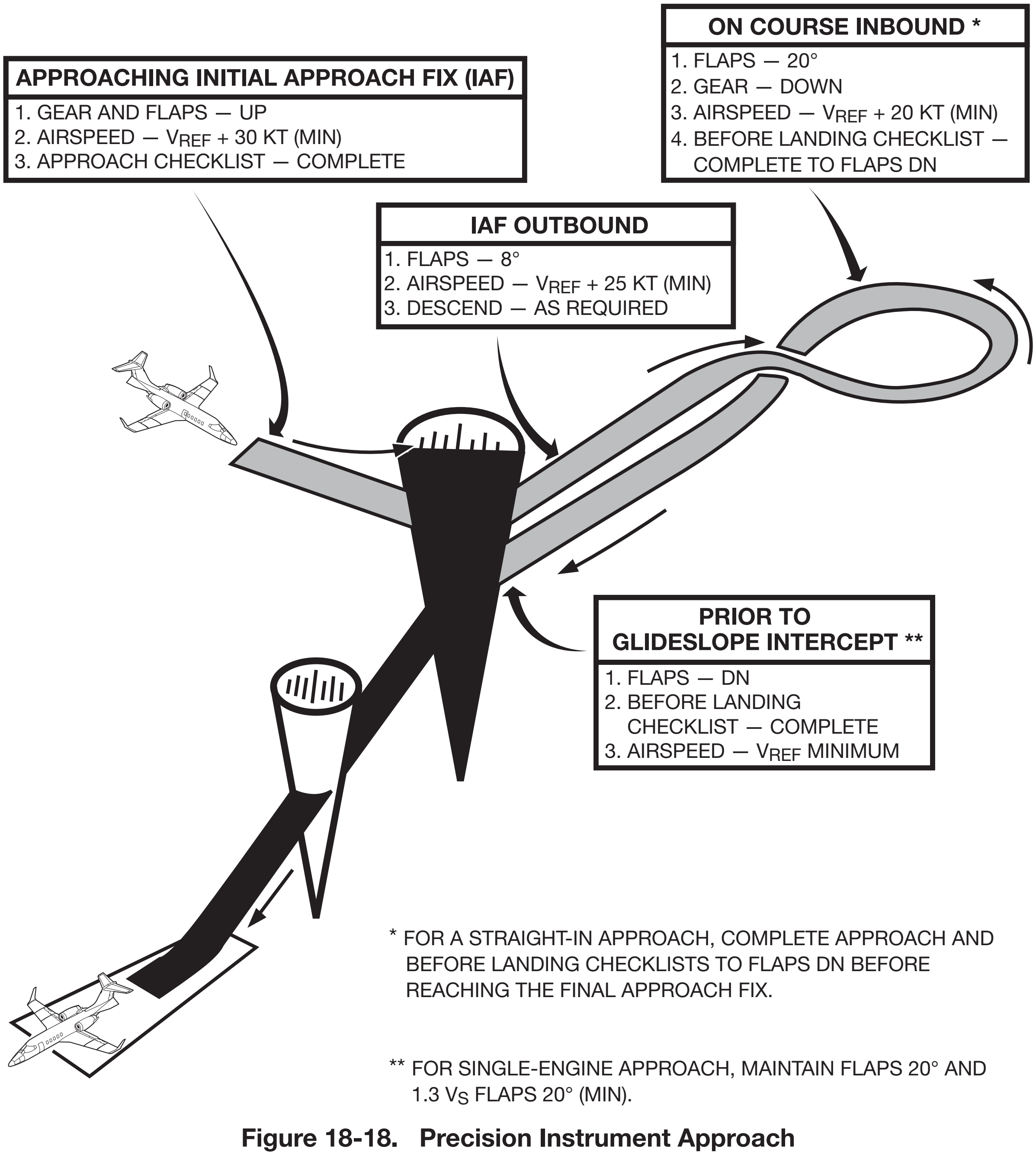


Figure 18-18. Precision Instrument Approach

NONPRECISION INSTRUMENT APPROACH

A typical nonprecision instrument approach is shown in Figure 18-19. All accepted instrument flying procedures and techniques should be used while making instrument approaches in the Learjet.

Two-engine, nonprecision approaches should be flown with a stabilized airspeed and configuration from the FAF inbound. Single-engine, nonprecision approaches should be flown at flaps 20° abnormal landings speed from the FAF inbound.

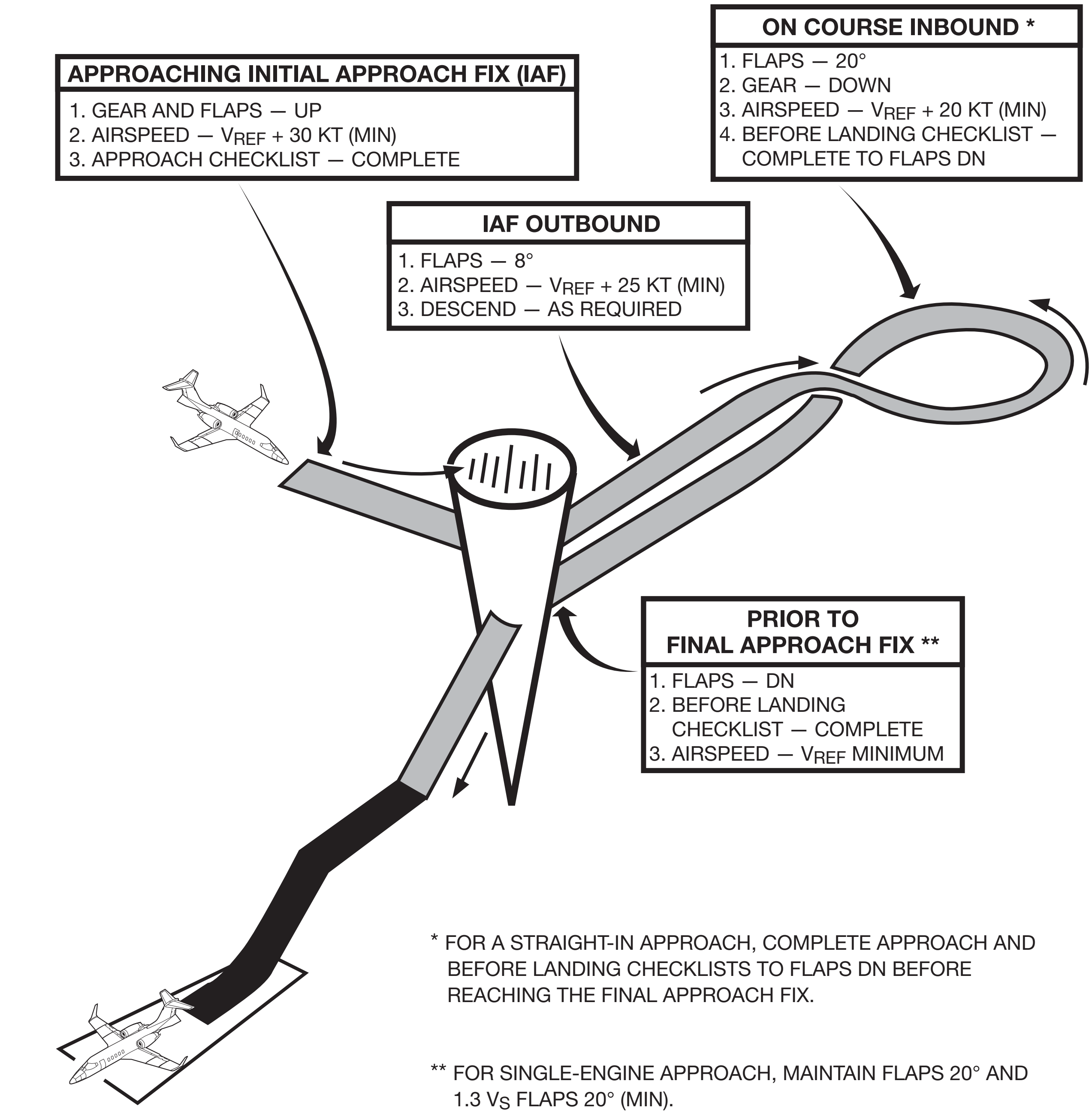


Figure 18-19. Nonprecision Instrument Approach

CIRCLING INSTRUMENT APPROACH

Any instrument approach that requires a heading change of 30° or more to line up with the landing runway is a circling approach. An identifiable part of the airport must be distinctly visible to the pilot during the circling approach, unless the inability to see an identifiable part of the airport results only from a normal bank of the airplane. The circling MDA and weather minima to be used are those for the runway to which the approach is flown.

The Learjet is an approach category C airplane. However, category D minimums must be used if the airplane will be maneuvered at speeds above 140 knots (the maximum for category C airplanes) during the circling approach.

There are two types of circling approaches. The first type of circling approach positions the airplane within 90°, or less, of the runway heading on a base leg for landing. With two engines, this type of approach is normally flown with the gear down and flaps DN at V_{REF} plus 10 knots from the FAF inbound. When landing is assured, airspeed may be reduced to V_{REF} minimum.

The second type of circling approach (Figure 18-20) requires a heading change of more than 90° to line up with the landing runway. With two engines, this type of approach is normally flown with the gear down and flaps DN at V_{REF} plus 10 knots from the FAF inbound. On final approach, reduce airspeed to V_{REF} minimum.

All single-engine circling approaches should be flown with flaps 20° at abnormal landing speed plus 10 knots from the FAF inbound.

NOTES

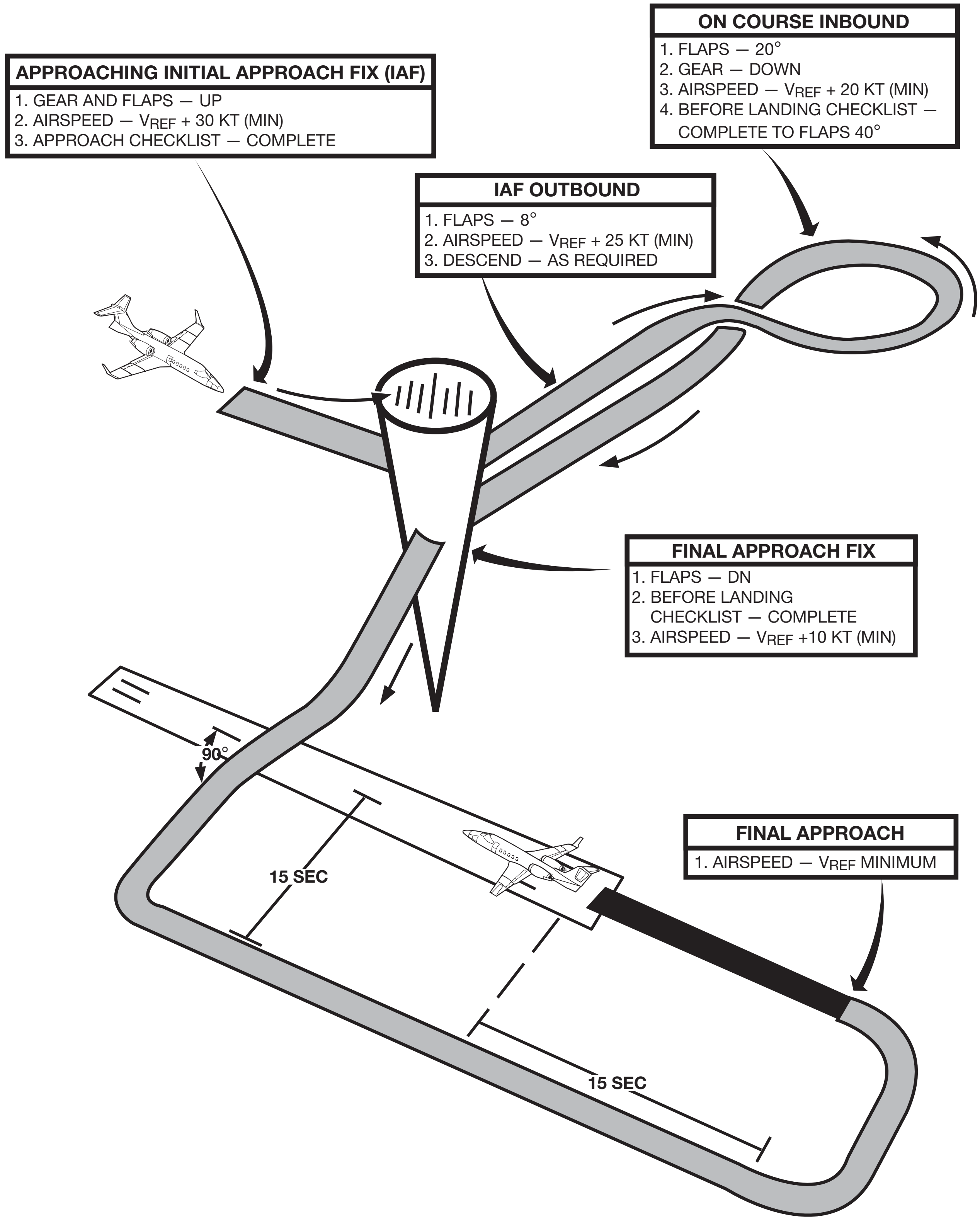


Figure 18-20. Circling Instrument Approach

GO-AROUND/BALKED LANDING

The Learjet go-around/balked landing procedure, shown in Figure 18-21, should be used for all missed approaches. Generally, if a missed approach is started at or above MDA or DH, it is considered a go-around. If a missed approach is started at or below MDA or DH, it is considered a rejected landing. During training, rejected landings will normally be initiated over the runway threshold at an altitude of approximately 50 feet.

In either case, use of the flight director go-around mode recommended to provide a target 9° nose-high pitch attitude and power may be adjusted to maintain the desired airspeed.

If the go-around/balked landing is made from an instrument approach, the published missed approach procedure should be accomplished unless otherwise instructed. If the go-around/balked landing is made during a circling approach, the initial turn to the missed approach heading must be made toward the landing runway. The turn may then be continued until the airplane is established on the missed approach heading.

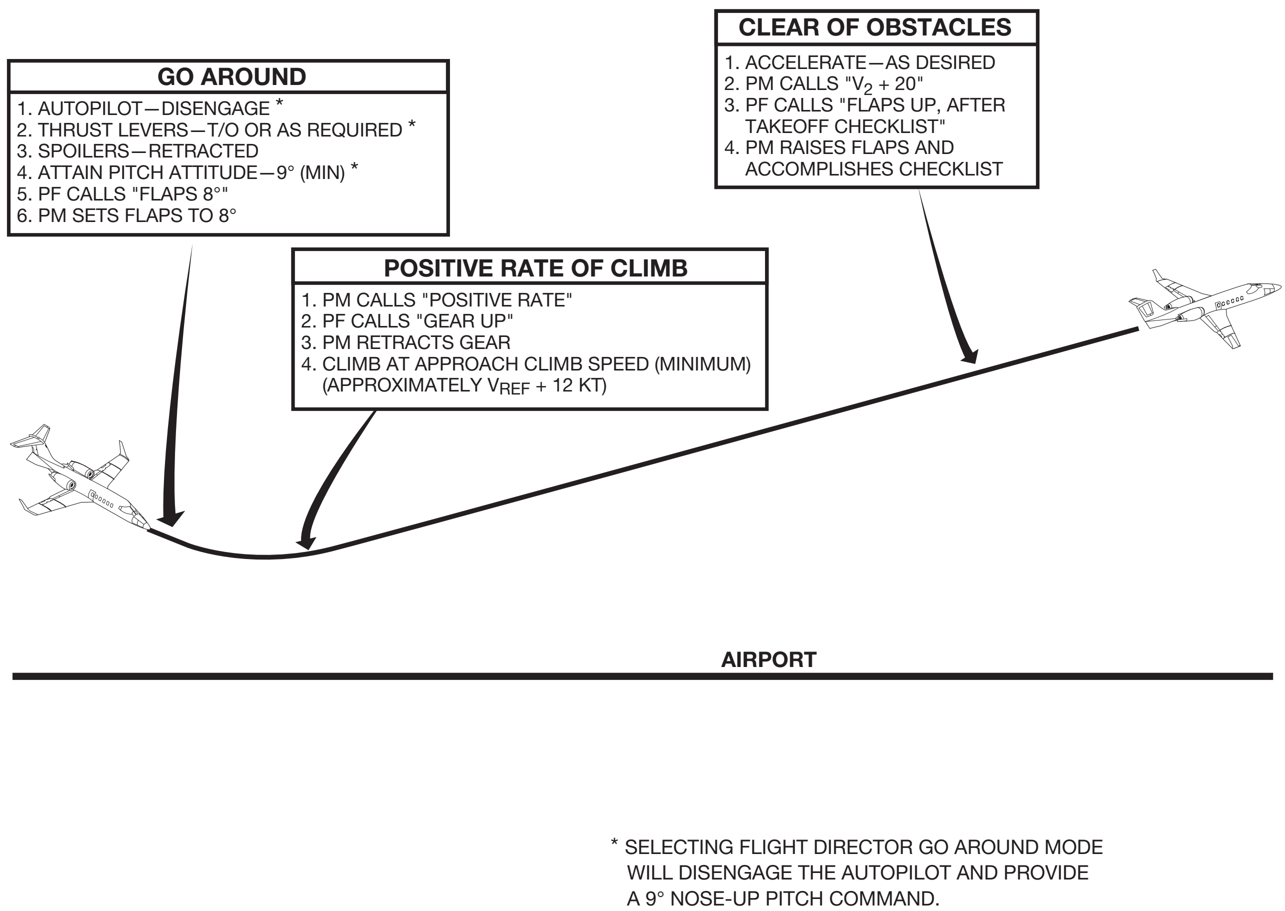
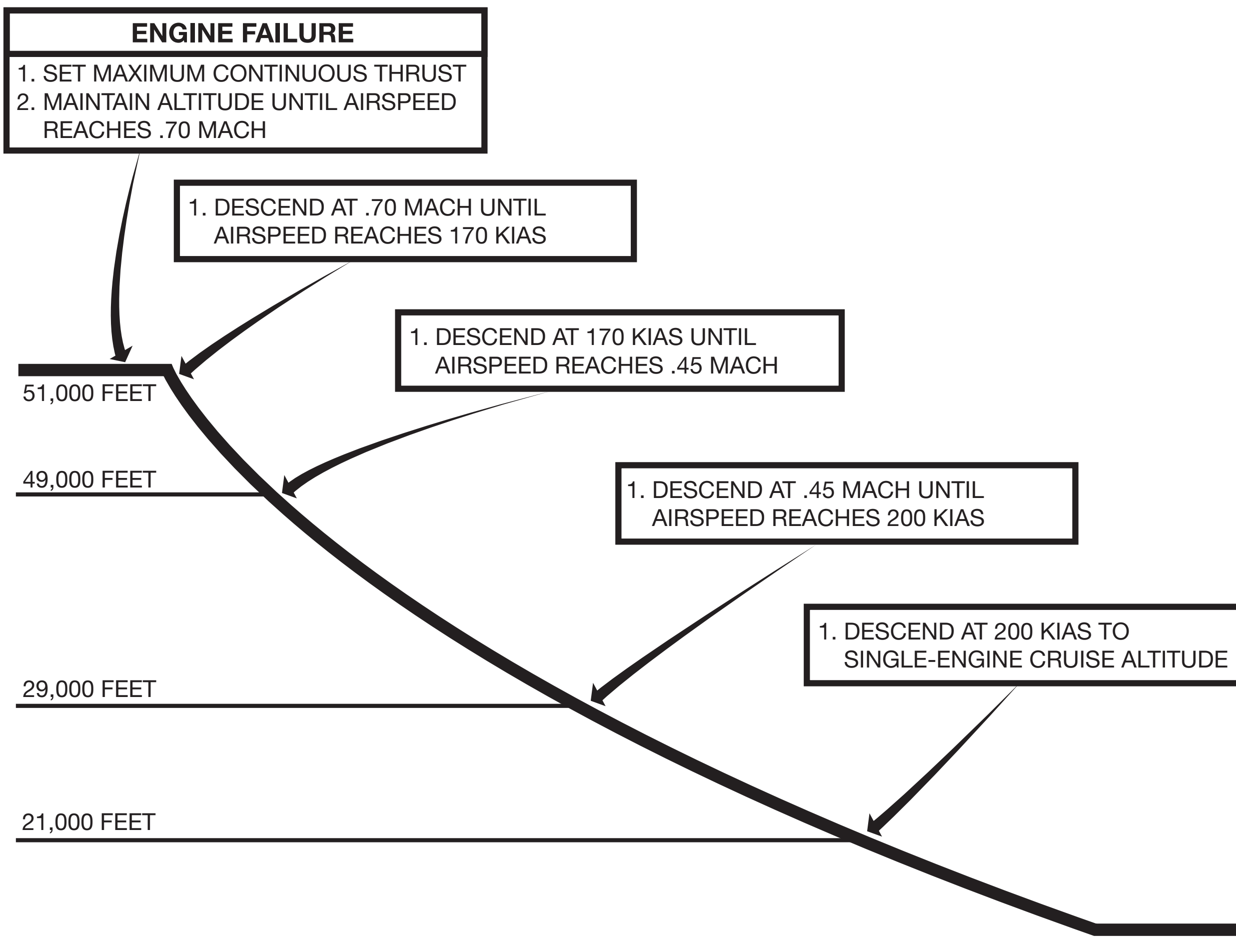


Figure 18-21. Go-Around/Balked Landing

SINGLE-ENGINE DRIFTDOWN

The single-engine driftdown procedure shown in Figure 18-22 is used to cover the greatest possible distance while descending to single-engine cruise altitude after an engine failure at high altitude.

As the note on the chart explains, the speed schedule depicted also approximates the best single-engine, rate-of-climb speed below the single-engine service ceiling. This speed schedule may then also be used to climb to single-engine cruise altitude after an engine failure at low altitude.



NOTE:

THIS SPEED SCHEDULE REPRESENTS THE MINIMUM SINK-RATE SPEED ABOVE THE SINGLE-ENGINE SERVICE CEILING AND APPROXIMATES THE BEST RATE-OF-CLIMB SPEED BELOW THE SINGLE-ENGINE SERVICE CEILING.

REFER TO LEARJET 60 AFM, PAGE 5-82 OR CHECKLIST P-29.

Figure 18-22. Single-Engine Driftdown

CHAPTER 19
WEIGHT AND BALANCE

CONTENTS

ILLUSTRATIONS

CHAPTER 19 WEIGHT AND BALANCE

INTRODUCTION

This section contains data and procedures enabling the aircraft operator to maintain weight and balance of the aircraft within the prescribed envelope. It is the responsibility of the operator to ensure that the aircraft is loaded properly. A separate data package, which is specific to a particular aircraft serial number, is provided by the manufacturer at time of initial aircraft weighing record, and a payload moments chart. It is the responsibility of the aircraft owner and operator to update this data package when required by aircraft changes.

Owners are advised to contact the aircraft manufacturer when any change is made to the aircraft which would appreciably affect the location of useful load items.

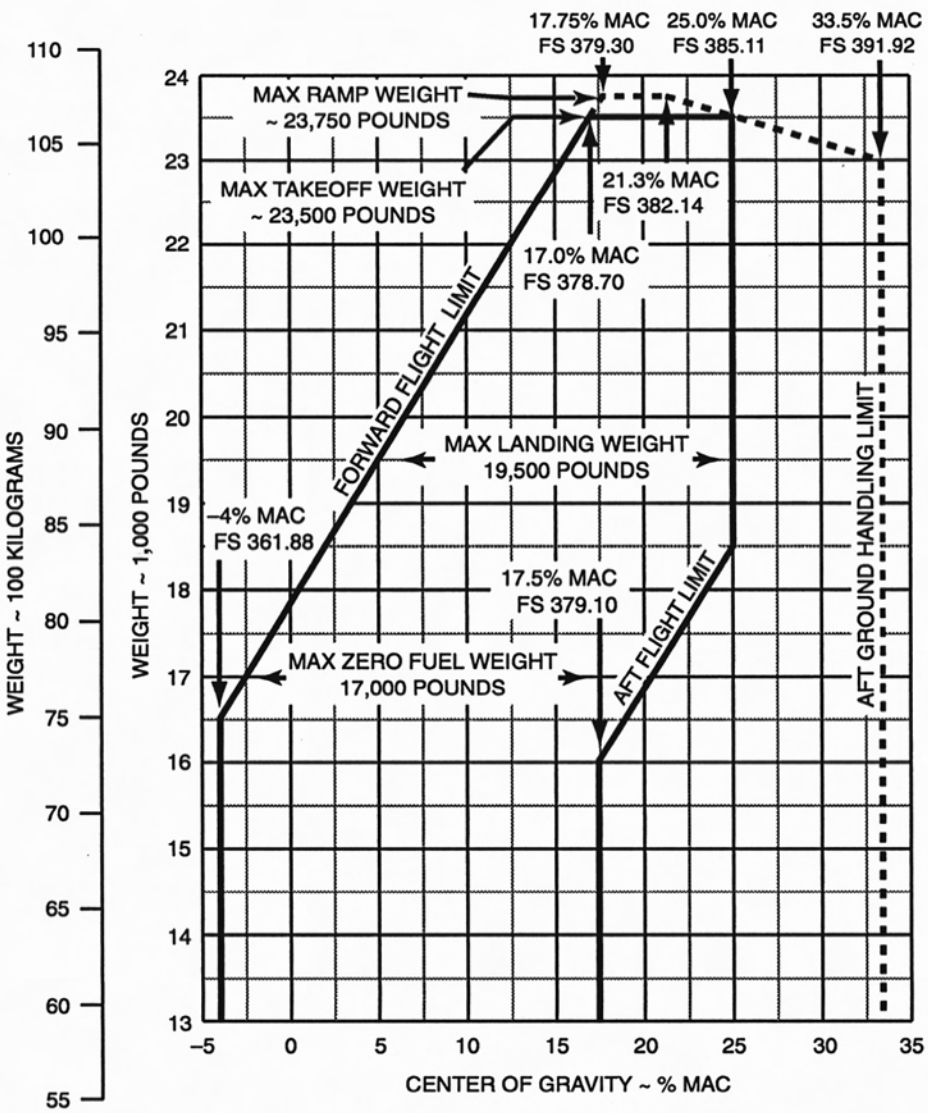
WEIGHT AND CENTER-OF-GRAVITY LIMITS

NOTE

The normal empty weight center of gravity may be aft of the flight limit.

The center of gravity of the airplane for all flight and ground conditions must be maintained within the applicable center-of-gravity envelope defined in Figure 19-1. (For dimensional data, refer to Figure 19-2.)

NOTES

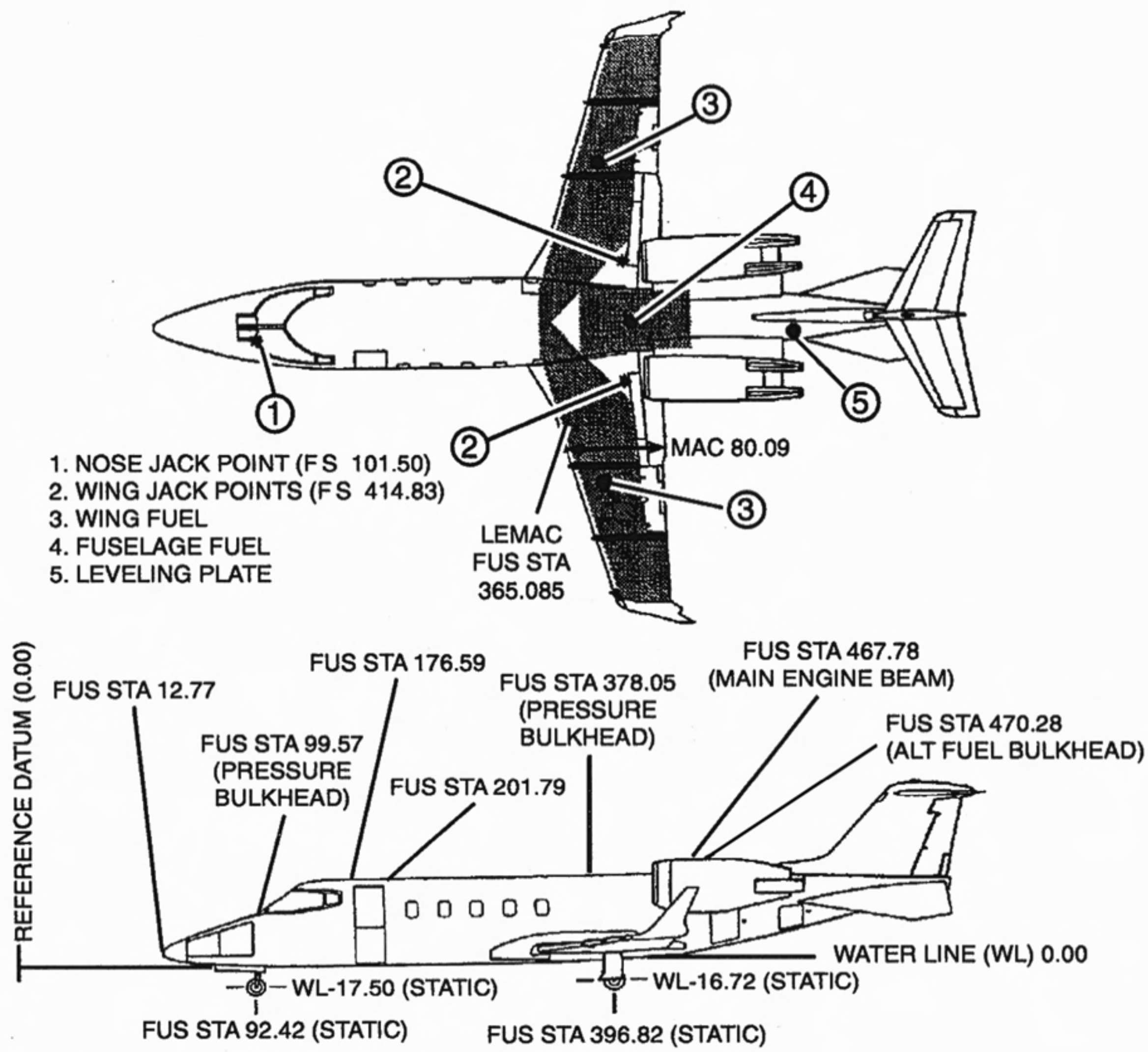


FORWARD FLIGHT LIMIT—F S 361.88 (–4.0% MAC) FOR ALL WEIGHTS UP TO AND INCLUDING 16,500 POUNDS (7,484 KG) AND TAPERS TO F S 378.70 (17.0% MAC) AT 23,500 POUNDS (10,660 KG).

AFT FLIGHT LIMIT—F S 379.10 (17.5% MAC) FOR ALL WEIGHTS UP TO AND INCLUDING 16,000 POUNDS (7,258 KG), TAPERS TO F S 385.11 (25.0% MAC) AT 18,500 POUNDS (8,392 KG), REMAINS AT F S 385.11 (25.0% MAC) UP TO AND INCLUDING 23,500 POUNDS (10,660 KG).

GROUND HANDLING LIMIT—THE FORWARD LIMIT IS THE SAME AS THE FORWARD FLIGHT LIMIT UP TO AND INCLUDING 23,500 POUNDS (10,660 KG) AND TAPERS TO F S 379.30 (17.75% MAC) AT 23,750 POUNDS (10,773 KG). THE AFT LIMIT IS F S 391.92 (33.5% MAC) FOR ALL WEIGHTS UP TO AND INCLUDING 22,987 POUNDS (10,427 KG) AND TAPERS TO F S 382.14 (21.3% MAC) AT 23,750 POUNDS (10,773 KG).

Figure 19-1. Center-of-Gravity Envelope—Aircraft with Optional 23,500 lb (10,660 kg) Takeoff Weight



NOSEWHEEL	GL WHEEL FUS STA	MAIN WHEEL	GL WHEEL FUS STA
FULLY COMPRESSED	92.72	FULLY COMPRESSED	397.00
STATIC	92.42	STATIC	396.82
FULLY EXTENDED	91.36	FULLY EXTENDED	396.04

Figure 19-2. Dimensional Data

LINEAR CONVERSIONS

Refer to Figure 19-3.

- To convert from centimeters to inches, find, in the bold face columns, the number of centimeters to be converted. The equivalent number of inches is read in the adjacent column headed “INCHES.”
- To convert from inches to centimeters, find, in the bold face columns, the number of inches to be converted. The equivalent number of centimeters is read in the adjacent column headed “CENTIMETERS.”

CENTIMETERS	◄►	INCHES	CENTIMETERS	◄►	INCHES
2.54	1	0.39	939.80	370	145.67
5.08	2	0.79	965.20	380	149.61
7.62	3	1.18	990.60	390	153.54
10.16	4	1.57	1016.00	400	157.48
12.70	5	1.97	1041.40	410	161.42
15.25	6	2.36	4066.80	420	165.35
17.78	7	2.76	1092.20	430	469.29
20.32	8	3.15	1117.60	440	173.23
22.86	9	3.54	1143.00	450	177.16
25.40	10	3.94	1168.40	460	181.10
50.80	20	7.87	1193.80	470	185.04
76.20	30	11.81	1216.20	480	188.98
101.60	40	15.75	1244.60	490	192.91
127.00	50	19.68	1270.00	500	196.85
152.40	60	23.62	1295.40	510	200.79
177.80	70	27.56	1320.80	520	204.72
203.20	80	31.50	1346.20	530	208.66
228.60	90	35.43	1371.60	540	212.60
254.00	100	39.37	1397.00	550	216.53
279.40	110	43.31	1422.40	560	220.47
304.80	120	47.24	1447.80	570	224.41
330.20	130	51.18	1473.20	580	228.35
355.60	140	55.12	1498.60	590	232.28
381.00	150	59.05	1524.00	600	236.22
406.40	160	62.99	1549.40	610	240.16
431.80	170	66.93	1574.80	620	244.09
457.20	180	70.87	1600.20	630	248.03
482.60	190	74.80	1625.60	640	251.97
508.00	200	78.74	1651.00	650	255.90
533.40	210	82.68	1676.40	660	259.84
558.80	220	86.61	1701.80	670	263.78
584.20	230	90.55	1727.20	680	267.72
609.60	240	94.49	1752.60	690	271.65
635.00	250	98.42	1778.00	700	275.59
660.40	260	102.36	1803.40	710	279.53
685.80	270	106.30	1828.80	720	283.46
711.20	280	110.24	1854.20	730	287.40
736.60	290	114.17	1879.60	740	291.34
762.00	300	118.11	1905.00	750	295.27
787.40	310	122.05	1930.40	760	299.21
812.80	320	125.98	1955.80	770	303.15
838.20	330	129.92	1981.20	780	307.09
863.60	340	133.86	2006.60	790	311.02
889.00	350	137.79	2032.00	800	314.96
914.40	360	141.73			

Figure 19-3. Linear Conversions

19 WEIGHT AND BALANCE

VOLUME CONVERSIONS

Refer to Figure 19-4.

- To convert from liters to gallons, find, in the bold face columns, the number of liters to be converted. The equivalent number of gallons is read in the adjacent column headed "GAL."
- To convert from gallons to liters, find, in the bold face columns, the number of gallons to be converted. The equivalent number of liters is read in the adjacent column headed "LITERS."

LITERS	◄►	GAL.	LITERS	◄►	GAL.	LITERS	◄►	GAL.
18.9	5	1.3	1476.2	390	103.0	2952.3	780	206.1
37.9	10	2.6	1514.0	400	105.7	2990.2	790	208.7
75.7	20	5.3	1551.9	410	108.3	3028.0	800	211.4
113.6	30	7.9	1589.7	420	111.0	3065.9	810	214.0
151.4	40	10.6	1627.6	430	113.6	3103.7	820	216.6
189.3	50	13.2	1665.4	440	116.2	3141.6	830	219.3
227.1	60	15.9	1703.3	450	118.9	3179.4	840	221.9
265.0	70	18.5	1741.1	460	121.5	3217.3	850	224.6
302.8	80	21.1	1779.0	470	124.2	3255.1	860	227.2
340.7	90	23.8	1816.8	480	126.8	3293.0	870	229.9
378.5	100	26.4	1854.7	490	129.5	3330.8	880	232.5
416.4	110	29.1	1892.5	500	132.1	3368.7	890	235.1
454.2	120	31.7	1930.4	510	134.7	3406.5	900	237.8
492.1	130	34.3	1968.2	520	137.4	3444.4	910	240.4
539.9	140	37.0	2006.1	530	140.0	3482.2	920	243.1
567.8	150	39.6	2043.9	540	142.7	3520.1	930	245.7
605.6	160	42.3	2081.8	550	145.3	3557.9	940	248.3
643.5	170	44.9	2119.6	560	148.0	3595.8	950	251.0
681.3	180	47.6	2157.5	570	150.6	3633.6	960	253.6
719.2	190	50.2	2195.3	580	153.2	3671.5	970	256.3
757.0	200	52.8	2233.2	590	155.9	3709.3	980	258.9
794.9	210	55.5	2271.0	600	158.5	3747.2	990	261.6
832.7	220	58.1	2308.9	610	161.2	3785.0	1000	264.2
870.6	230	60.8	2346.7	620	163.8	4163.5	1100	290.6
908.4	240	63.4	2384.6	630	166.4	4542.0	1200	317.0
946.3	250	66.1	2422.4	640	169.1	4920.5	1300	343.5
984.1	260	68.7	2460.3	650	171.7	5299.0	1400	369.9
1022.0	270	71.3	2498.1	660	174.4	5677.5	1500	396.3
1059.8	280	74.0	2536.0	670	177.0	6056.0	1600	422.7
1097.7	290	76.6	2573.8	680	179.7	6434.5	1700	449.1
1135.5	300	79.3	2611.7	690	182.3	6813.0	1800	475.6
1173.4	310	81.9	2649.5	700	184.9	7191.5	1900	502.0
1211.2	320	84.5	2687.4	710	187.6	7570.0	2000	528.4
1249.1	330	87.2	2725.2	720	190.2	7948.5	2100	554.8
1286.9	340	89.8	2763.1	730	192.9	8327.0	2200	581.2
1324.8	350	92.5	2800.9	740	195.5	8705.5	2300	607.7
1362.6	360	95.1	2838.8	750	198.2	9084.0	2400	634.1
1400.5	370	97.8	2876.6	760	200.8	9462.5	2500	660.5
1438.3	380	100.4	2914.5	770	203.4	9841.0	2600	686.9

Figure 19-4. Volume Conversions

WEIGHT CONVERSIONS

Refer to Figure 19-5.

- To convert from kilograms to pounds, find, in the bold face columns, the number of kilograms to be converted. The equivalent number of pounds is read in the adjacent column headed "LB."
- To convert from pounds to kilograms, find, in the bold face columns, the number of pounds to be converted. The equivalent number of kilograms is read in the adjacent column headed "KG."

KG	◄►	LB	KG	◄►	LB	KG	◄►	LB
4.5	10	22.0	208.7	460	1014.1	412.8	910	2006.2
9.1	20	44.1	213.2	470	1036.2	417.3	920	2028.2
13.6	30	66.1	217.7	480	1058.2	421.8	930	2050.3
18.1	40	88.2	222.3	490	1080.3	426.4	940	2072.3
22.7	50	110.2	226.8	500	1102.3	430.9	950	2094.4
27.2	60	132.3	231.3	510	1124.3	435.5	960	2116.4
31.8	70	154.3	235.9	520	1146.4	440.0	970	2138.5
36.3	80	176.4	240.4	530	1168.4	444.5	980	2160.5
40.8	90	198.4	244.9	540	1190.5	449.1	990	2182.6
45.4	100	220.5	249.5	550	1212.5	453.6	1000	2204.6
49.9	110	242.5	254.0	560	1234.6	499.0	1100	2425.1
54.4	120	264.6	258.6	570	1256.6	544.3	1200	2645.5
59.0	130	286.6	263.1	580	1278.7	589.7	1300	2866.0
63.5	140	308.6	267.6	590	1300.7	635.0	1400	3086.4
68.0	150	330.7	272.2	600	1322.8	680.4	1500	3306.9
72.6	160	352.7	276.7	610	1344.8	907.2	2000	4409.2
77.1	170	374.8	281.2	620	1366.9	1134.0	2500	5511.5
81.6	180	396.8	285.8	630	1388.9	1360.8	3000	6613.8
86.2	190	418.9	290.3	640	1410.9	1587.6	3500	7716.1
90.7	200	440.9	294.8	650	1433.0	1814.4	4000	8818.4
95.3	210	463.0	299.4	660	1455.0	2041.2	4500	9920.7
99.8	220	485.0	303.9	670	1477.1	2268.0	5000	11023.0
104.3	230	507.1	308.4	680	1499.1	3494.8	5500	12125.3
108.9	240	529.0	313.0	690	1521.2	2721.6	6000	13227.6
113.4	250	551.1	317.5	700	1543.2	2948.4	6500	14329.9
117.9	260	573.2	322.1	710	1565.3	3175.2	7000	15432.2
122.5	270	595.2	326.6	720	1587.3	3402.0	7500	16534.5
127.0	280	617.3	331.1	730	1609.4	3628.8	8000	17636.8
131.5	290	639.3	335.7	740	1631.4	3855.6	8500	18739.1
136.1	300	661.4	340.2	750	1653.4	4082.4	9000	19841.4
140.6	310	683.4	344.7	760	1675.5	4309.2	9500	20943.7
145.2	320	705.5	349.3	770	1697.5	4536.0	10000	22046.0
149.7	330	727.5	353.8	780	1719.6	4989.6	11000	24250.6
154.2	340	749.6	358.3	790	1741.6	5443.2	12000	26455.2
158.8	350	771.6	362.9	800	1763.7	5896.8	13000	28659.8
163.3	360	793.7	367.4	810	1785.7	6350.4	14000	30864.4
167.8	370	815.7	371.9	820	1807.8	6804.0	15000	33069.0
172.4	380	837.7	376.5	830	1829.8	7257.6	16000	35273.6
176.9	390	859.8	381.0	840	1851.9	7711.1	17000	37478.2
181.4	400	881.8	385.6	850	1873.9	8164.7	18000	39682.8
186.0	410	903.9	390.1	860	1896.0	9618.3	19000	41887.4
190.5	420	925.9	394.6	870	1918.0	9071.9	20000	44092.0
195.0	430	948.0	399.2	880	1940.0	9525.5	21000	46296.6
199.6	440	970.0	403.7	890	1962.1	9979.1	22000	48501.2
204.1	450	992.1	408.2	900	1984.1	10432.7	23000	50705.8

Figure 19-5. Weight Conversions

LOADING

LOADING INSTRUCTIONS

It is the responsibility of the pilot to see that this aircraft is loaded within the weight and CG limits. The loading form (Figure 19-6) may be used.

1. Enter the aircraft's BASIC EMPTY WEIGHT and MOMENT from the current weighing record.
2. Enter the payload weights and moments (crew, passengers, provisions, baggage, etc.) using the ***Payload Moments*** charts provided in the aircraft's weight and balance data package.
3. Compute the ZERO FUEL WEIGHT and MOMENT (OPERATING WEIGHT values plus passenger and baggage values).
4. Enter the fuel weights and moments using the Fuel Moments chart, Figure 19-9.
5. Compute RAMP WEIGHT and MOMENT (ZERO FUEL WEIGHT values plus fuel values).
6. Compute TAKEOFF WEIGHT and MOMENT (RAMP WEIGHT values minus taxi burnoff out of wings).
7. Compare TAKEOFF WEIGHT and MOMENT with weight and CG limits from Weight-Moment-CG Envelope, Figure 19-7, or Center-of-Gravity Table, Figure 19-8. If not within limits, reduce weight or rearrange load as required to obtain weight and CG within limits.
8. LANDING WEIGHT and MOMENT may be calculated by adding the fuel weight and moment remaining at the destination to the ZERO FUEL WEIGHT.
9. The formula to calculate the CG in % MAC is:

$$\text{CG in \% MAC} = \left[\frac{\begin{array}{l}\text{Fuselage Station}\\ \text{(Center of Gravity)} - 365.085\end{array}}{80.09}\right] \times 100$$

NOTES

AIRCRAFT LOADING FORM

INTERIOR CONFIGURATION ______________________________

MISSING OR ADDITIONAL EQUIPMENT ______________________________

	WEIGHT	FS	MOM/1000	% MAC
BASIC EMPTY WEIGHT				
MISSING/ADDITIONAL EQUIPMENT				
CREW				
PROVISIONS—REFRESHMENT CABINET				
PROVISIONS—REFRESHMENT CABINET				
PROVISIONS—VANITY/LAVATORY				
PROVISIONS—TOILET				
WATER				
MISCELLANEOUS				
OPERATING WEIGHT EMPTY				
BAGGAGE—CABIN				
BAGGAGE—TAILCONE				
PASSENGER 1				
PASSENGER 2				
PASSENGER 3				
PASSENGER 4				
PASSENGER 5				
PASSENGER 6				
PASSENGER 7				
PASSENGER 8				
ZERO FUEL WEIGHT				
FUEL—FUSELAGE TANKS				
FUEL—WING TANKS				
RAMP WEIGHT				
TAXI BURNOFF OUT OF WINGS*				
TAKEOFF GROSS WEIGHT				

ZERO FUEL WEIGHT				
FUEL—WING TANKS				
FUEL—FUSELAGE TANKS				
LANDING WEIGHT				

*FUEL BURNOFF IS APPROXIMATELY 3.7 POUNDS PER ENGINE PER MINUTE

Figure 19-6. Aircraft Loading Form

NOTE

The landing gear retraction moment is (minus) 2,655 inch-pounds.

EFFECTIVITY

60-049 & Subsequent and prior aircraft incorporating SB 60-8-1

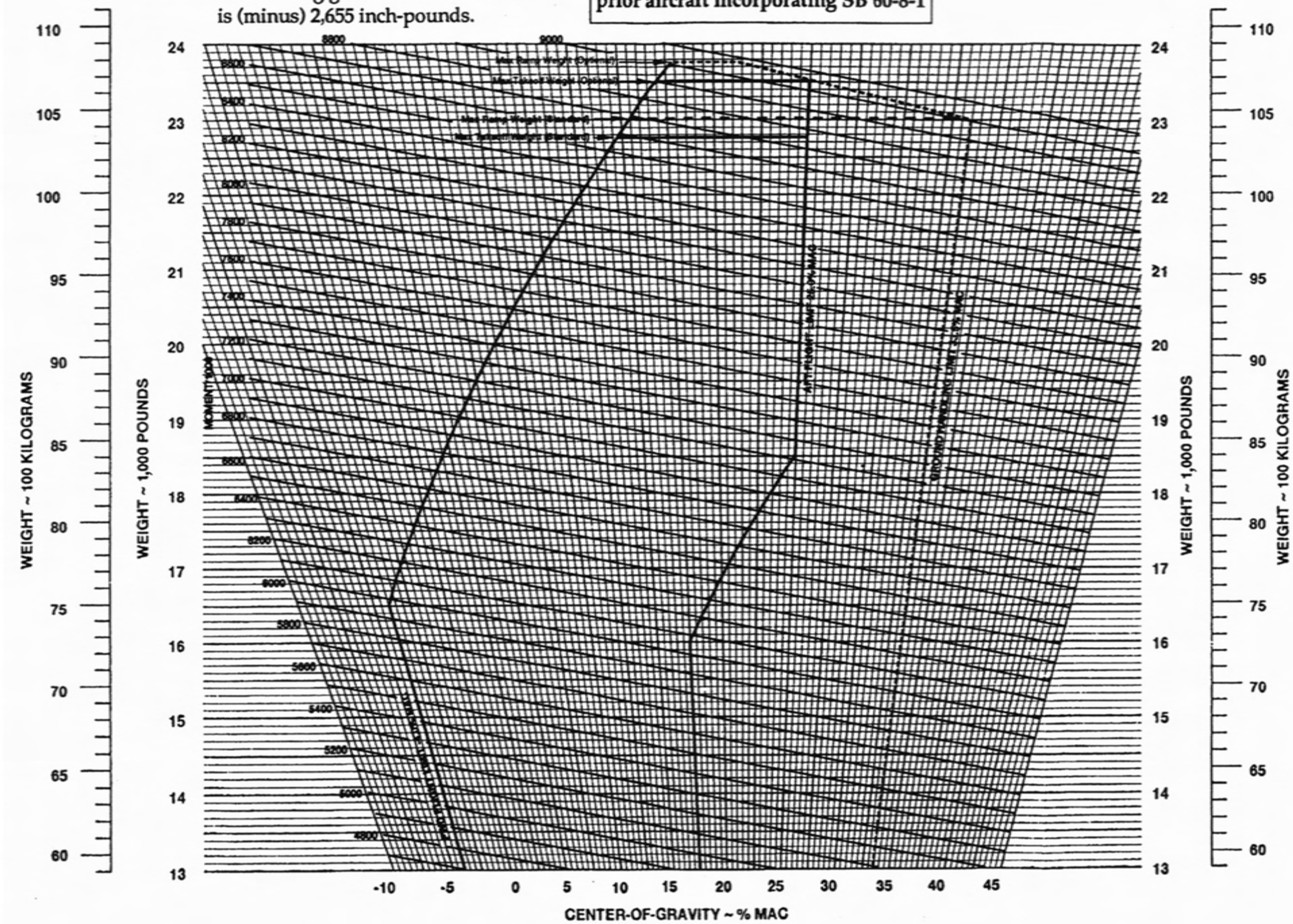


Figure 19-7. Weight-Moment-CG Envelope (Sheet 1 of 2)

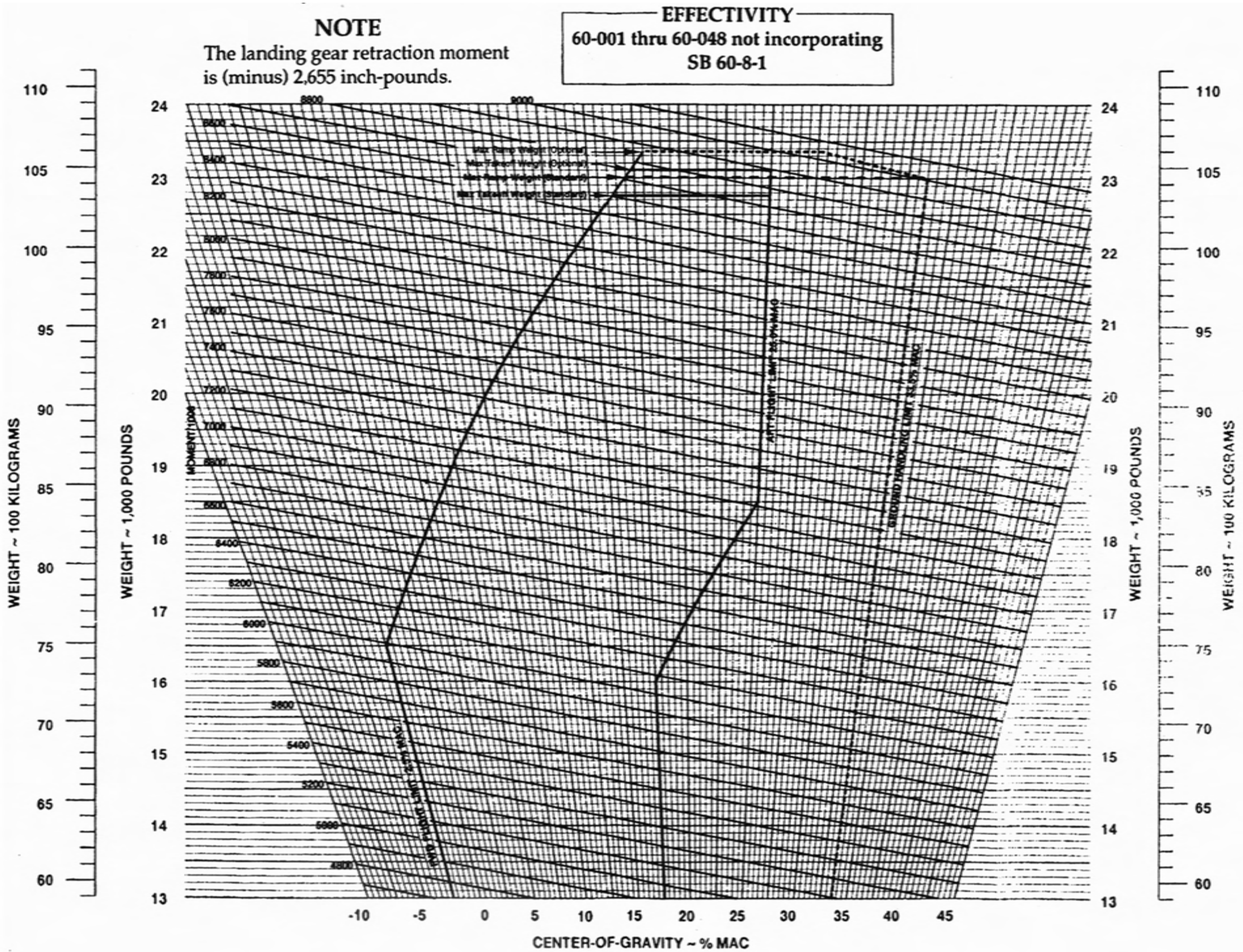


Figure 19-7. Weight-Moment-CG Envelope (Sheet 2 of 2)

NOTE:
DATA GIVEN FOR WEIGHTS ABOVE 23,500 POUNDS
REFLECTS THE GROUND HANDLING ENVELOPE ONLY.

MOMENT/1,000

POUNDS GROSS WIEGHT	FWD LIMIT %MAC	FWD LIMIT STA		0%MAC STA 365.08	5%MAC STA 369.09	10%MAC STA 373.09	15%MAC STA 377.10	20%MAC STA 381.10	AFT LIMIT %MAC	AFT LIMIT STA	
13000	–4.00	361.88	4704.48	4746.10	4798.16	4850.22	4902.28		17.50	379.10	4928.31
13100	–4.00	361.88	4740.65	4782.61	4835.07	4887.53	4939.99		17.50	379.10	4966.22
13200	–4.00	361.88	4776.83	4819.12	4871.98	4924.84	4977.70		17.50	379.10	5004.13
13300	–4.00	361.88	4813.02	4855.53	4908.89	4962.15	5015.41		17.50	379.10	5042.04
13400	–4.00	361.88	4849.21	4892.14	4945.80	4999.46	5053.12		17.50	379.10	5079.95
13500	–4.00	361.88	4885.40	4928.65	4982.71	5036.77	5090.83		17.50	379.10	5117.86
13600	–4.00	361.88	4921.59	4965.16	5019.82	5074.08	5128.54		17.50	379.10	5155.77
13700	–4.00	361.88	4957.78	5001.66	5056.53	5111.39	5166.25	BEYOND LIMITS	17.50	379.10	5193.68
13800	–4.00	361.88	4993.96	5038.17	5093.44	5148.70	5203.96		17.50	379.10	5231.59
13900	–4.00	361.88	5030.15	5074.68	5130.34	5186.01	5241.67		17.50	379.10	5269.50
14000	–4.00	361.88	5066.34	5111.19	5167.25	5223.32	5279.38		17.50	379.10	5307.41
14100	–4.00	361.88	5102.53	5147.70	5204.18	5260.63	5317.09		17.50	379.10	5345.32
14200	–4.00	361.88	5136.72	5184.21	5241.07	5297.93	5354.80		17.50	379.10	5383.23
14300	–4.00	361.88	5174.90	5220.72	5277.98	5335.24	5392.51		17.50	379.10	5421.14
14400	–4.00	361.88	5211.09	5257.22	5314.89	5372.55	5430.22		17.50	379.10	5459.05
14500	–4.00	361.88	5247.26	5293.73	5351.80	5409.86	5467.93		17.50	379.10	5496.96
14600	–4.00	361.88	5238.47	5330.24	5388.71	5447.17	5505.64		17.50	379.10	5534.87
14700	–4.00	361.88	5319.66	5366.75	5425.62	5484.48	5543.35		17.50	379.10	5572.78
14800	–4.00	361.88	5355.84	5403.26	5462.52	5521.79	5581.06		17.50	379.10	5610.69
14900	–4.00	361.88	5392.03	5439.77	5499.43	5559.10	5618.77		17.50	379.10	5648.60
15000	–4.00	361.88	5428.22	5476.27	5536.34	5596.41	5656.48		17.50	379.10	5686.51
15100	–4.00	361.88	5464.41	5512.78	5573.25	5633.72	5694.19		17.50	379.10	5724.42
15200	–4.00	361.88	5500.60	5549.29	5610.16	5671.03	5731.90		17.50	379.10	5762.33
15300	–4.00	361.88	5536.79	5585.80	5647.07	5708.34	5769.61		17.50	379.10	5800.24
15400	–4.00	361.88	5572.97	5622.31	5683.80	5745.65	5807.32		17.50	379.10	5838.15
15500	–4.00	361.88	5609.16	5658.82	5720.89	5782.96	5845.03		17.50	379.10	5876.06
15600	–4.00	361.88	5645.35	5695.33	5757.80	5820.27	5882.74		17.50	379.10	5913.97
15700	–4.00	361.88	5681.54	5731.83	5794.71	5857.58	5920.74		17.50	379.10	5951.88
15800	–4.00	361.88	5717.73	5768.34	5831.61	5894.89	5958.16		17.50	379.10	5989.79
15900	–4.00	361.88	5753.91	5804.85	5868.52	5932.19	5995.87		17.50	379.10	6027.70
16000	–4.00	361.88	5790.10	5841.36	5905.43	5969.50	6033.58		17.50	379.10	6065.61
16100	–4.00	361.88	5826.29	5877.87	5942.34	6006.81	6071.29		17.80	379.34	6107.39
16200	–4.00	361.88	5882.48	5914.38	5979.25	6044.12	6109.00		18.10	379.58	6149.22
16300	–4.00	361.88	5898.67	5950.89	6016.16	6081.43	6146.71		18.40	379.82	6191.09
16400	–4.00	361.88	5934.85	5987.39	6053.07	6118.74	6184.42		18.70	380.06	6233.01
16500	–4.00	361.88	5971.04	6023.90	6089.98	6156.05	6222.13		19.00	380.30	6274.96
16600	–3.70	362.12	6011.22	6060.41	6126.89	6193.36	6259.84		19.30	380.54	6317.00
16700	–3.40	362.36	6051.44	6096.92	6163.79	6230.67	6297.54		19.60	380.78	6359.07
16800	–3.10	362.60	6091.72	6133.43	6200.70	6287.96	6335.25		19.90	381.02	6401.18
16900	–2.80	362.84	6132.04	6169.94	6237.61	6305.29	6372.96	6440.64	20.20	381.26	6443.35
17000	–2.50	363.80	6172.41	6206.44	6274.52	6342.80	6410.67	6478.75	20.50	381.50	6485.56
17100	–2.20	363.32	6212.82	6242.95	6311.43	6379.91	6448.38	6516.86	20.80	381.74	6527.82
17200	–1.90	363.56	6253.29	6279.46	6348.34	6417.22	6486.09	6554.97	21.10	681.96	6570.12
17300	–1.60	363.80	6293.80	6315.97	6385.25	6454.53	6523.80	6593.08	21.40	682.22	6612.48
17400	–1.30	364.04	6334.36	6352.48	3422.16	6491.84	6581.51	6631.19	21.70	382.45	6654.88
17500	–1.00	364.28	6374.97	6388.99	6459.07	6529.14	6599.22	6669.30	22.00	382.70	6697.33
17600	–0.70	364.52	6415.83	6425.50	6495.98	6566.45	6636.93	6707.41	22.30	382.95	6739.83
17700	–0.40	364.78	6456.33	6482.00	6532.88	6603.76	6674.64	6745.52	22.60	383.19	6782.38
17800	–0.10	365.00	6497.09	6498.51	6569.79	6641.07	6712.35	6783.63	22.90	383.43	6824.96
17900	0.20	365.25	6537.89		6806.70	6678.36	6750.06	6821.74	23.20	383.67	6867.62
18000	0.50	365.49	6578.74		6715.69	6715.69	6787.77	6859.85	23.50	383.91	6910.31
18100	0.80	365.73	6619.64		6753.00	6753.00	6825.46	6897.96	23.80	384.15	6953.05
18200	1.10	365.97	6660.58		6790.31	6790.31	6863.19	6936.07	24.10	384.39	6995.84
18300	1.40	366.21	6701.57		6827.62	6827.62	6900.90	6974.18	24.40	384.63	7038.67
18400	1.70	366.45	6742.62		6864.93	6864.93	6938.61	7012.30	24.70	384.87	7081.56

Figure 19-8. Center-of-Gravity Table (Sheet 1 of 2)

NOTE:
DATA GIVEN FOR WEIGHTS ABOVE 23,500 POUNDS
REFLECTS THE GROUND HANDLING ENVELOPE ONLY.

MOMENT/1,000

POUNDS GROSS WIEGHT	FWD LIMIT %MAC	FWD LIMIT STA		5%MAC STA 369.09	10%MAC STA 373.09	15%MAC STA 377.10	20%MAC STA 381.10	AFT LIMIT %MAC	AFT LIMIT STA	
18500	2.00	366.69	6783.71	6828.16	6902.24	6976.32	7050.41	25.00	385.11	7124.49
18600	2.30	366.93	6824.84	6885.06	6939.55	7014.03	7088.52	25.00	385.11	7163.00
18700	2.60	367.17	6866.03	6901.97	6976.86	7051.74	7126.63	25.00	385.11	7201.51
18800	2.90	367.41	6907.26	6938.88	7014.17	7089.45	7164.74	25.00	385.11	7240.02
18900	3.20	367.65	6948.54	6975.79	7051.48	7127.16	7202.85	25.00	385.11	7278.53
19000	3.50	367.89	6989.87	7012.70	7088.79	7164.87	7240.96	25.00	385.11	7317.04
19100	3.80	368.13	7031.25	7049.61	7126.10	7202.58	7279.07	25.00	385.11	7355.55
19200	4.10	368.37	7072.68	7086.52	7163.40	7240.29	7317.18	25.00	385.11	7394.06
19300	4.40	368.61	7114.15	7123.43	7200.71	7278.00	7355.29	25.00	385.11	7432.57
19400	4.70	368.85	7155.68	7160.34	7238.02	7315.71	7393.40	25.00	385.11	7471.09
19500	5.00	369.09	7197.25	7197.25	7275.33	7353.42	7431.51	25.00	385.11	7509.60
19600	5.30	369.33	7238.86		7312.64	7397.13	7469.62	25.00	385.11	7548.11
19700	5.60	369.57	7280.53		7349.95	7428.84	7507.73	25.00	385.11	7586.62
19800	5.90	369.81	7322.24		7387.26	7466.55	7545.84	25.00	385.11	7625.13
19900	6.20	370.05	7364.01		7424.57	7504.26	7583.95	25.00	385.11	7663.64
20000	6.50	370.29	7405.82		7461.88	7541.97	7622.06	25.00	385.11	7702.15
20100	6.80	370.53	7447.68		7499.19	7579.68	7660.17	25.00	385.11	7740.66
20200	7.10	370.77	7489.58		7536.50	7617.39	7698.28	25.00	385.11	7779.17
20300	7.40	371.01	7531.54		7573.81	7655.10	7736.39	25.00	385.11	7817.68
20400	7.70	371.25	7573.54		7611.12	7692.81	7774.50	25.00	385.11	7856.19
20500	8.00	371.49	7615.59		7648.43	7730.52	7812.61	25.00	385.11	7894.70
20600	8.30	371.73	7657.69		7685.74	7768.23	7850.72	25.00	385.11	7933.21
20700	8.60	371.97	7699.84		7723.05	7805.94	7888.83	25.00	385.11	7971.73
20800	8.90	372.21	7742.03		7760.36	7843.65	7926.94	25.00	385.11	8010.24
20900	9.20	372.45	7784.27		7797.66	7881.36	7965.05	25.00	385.11	8048.75
21000	9.50	372.69	7826.56		7834.97	7919.07	8003.16	25.00	385.11	8087.26
21100	9.80	372.93	7868.90		7872.28	7658.78	8041.27	25.00	385.11	8125.77
21200	10.10	373.17	7911.29			7994.49	8079.38	25.00	385.11	8164.28
21300	10.40	373.41	7953.73			8032.20	8117.49	25.00	385.11	8202.79
21400	10.70	373.65	7996.21			8069.91	8155.60	25.00	385.11	8241.30
21500	11.00	373.89	8029.74			8107.62	8193.71	25.00	385.11	8279.81
21600	11.30	374.14	8061.32			8145.33	8231.82	25.00	385.11	8318.32
21700	11.60	374.38	8123.95			8183.04	8269.94	25.00	385.11	8356.83
21800	11.90	374.62	8166.62			8220.75	8308.05	25.00	385.11	8395.34
21900	12.20	374.86	8209.35			8258.46	8486.16	25.00	385.11	8433.85
22000	12.50	375.10	8252.12			8296.17	8384.27	25.00	385.11	8472.36
22100	12.80	375.34	8294.94			8333.88	8422.38	25.00	385.11	8510.88
22200	13.10	375.58	8337.80			8371.59	8460.49	25.00	385.11	8549.39
22300	13.40	375.82	8380.72			8409.30	8498.60	25.00	385.11	8587.90
22400	13.70	376.06	8423.68			8447.01	8536.71	25.00	385.11	8626.41
22500	14.00	376.30	8466.70			8484.72	8574.82	25.00	385.11	8664.92
22600	14.30	376.54	8509.76	BEYOND		8522.43	8612.93	25.00	385.11	8703.43
22700	14.60	376.78	8552.86	LIMITS		8560.14	8651.04	25.00	385.11	8741.94
22800	14.90	377.02	8596.02			8597.85	8689.15	25.00	385.11	8780.45
22900	15.20	377.26	8639.22				8727.26	25.00	385.11	8818.96
23000	15.50	377.50	8682.48				8765.37	25.00	385.11	8857.47
23100	15.80	377.74	8725.78				8803.48	25.00	385.11	8895.98
23200	16.10	377.98	8769.12				8841.59	25.00	385.11	8934.49
23300	16.40	378.22	8812.52				8879.70	25.00	385.11	8973.00
23400	16.70	378.46	8855.96				8917.81	25.00	385.11	9011.52
23500	17.00	378.70	8899.46				8955.92	25.00	385.11	9050.03
23600	17.30	378.94	8646.00				8894.03	23.70	384.07	9063.94
23700	17.60	379.18	8986.59				9032.14	22.10	382.78	9071.99
23750	17.75	379.30	9008.40				9051.20	21.30	382.14	9075.92

Figure 19-8. Center-of-Gravity Table (Sheet 2 of 2)

WEIGHING INSTRUCTIONS

Occasional weighing may be required to keep the basic empty weight current. All changes to the aircraft affecting weight and balance are the responsibility of the aircraft operator. For additional detailed information relating to weighing and leveling procedures, refer to *Maintenance Manual*, Chapter 8.

1. The basic empty weight CG is established with the wheels down.

2. Fuel should be drained through the drain valves prior to weighing and with the aircraft in level or static position. If unable to drain fuel, refer to the Unusable Fuel Table to determine weight and balance. Unusable and trapped fluids (Figure 19-9) are included in the basic empty weight.

3. The engine oil must be at full level in each oil tank. Total engine oil is 34 pounds at fuselage station 467.1.

4. Hydraulic reservoir, accumulator, oxygen bottles, alcohol tank, and gear shock struts must be filled to normal operating capacities.

5. Determine aircraft configuration at time of weighing. Missing items or items in aircraft (but not part of basic empty weight) should be noted and the final

POUNDS/KG	GALLONS/LITERS KEROSENE 6.7 LB/GAL	MOMENT/1,000 WING KEROSENE	MOMENT/1,000 FUSELAGE KEROSENE
100/ 45	**15/** 56	38.41	44.55
200/ 90	**30/** 113	76.85	89.18
300/ 136	**45/** 169	114.75	133.67
400/ 181	**60/** 226	154.42	178.07
500/ 226	**75/** 282	190.05	222.64
600/ 272	**90/** 339	228.02	267.39
700/ 317	**104/** 395	266.33	312.18
800/ 362	**119/** 452	304.70	356.51
900/ 408	**134/** 508	343.13	400.17
1000/ 453	**149/** 565	381.71	443.06
1100/ 499	**164/** 621	420.36	485.48
1200/ 544	**179/** 678	459.03	527.86
1300/ 589	**194/** 734	497.87	570.23
1400/ 635	**209/** 791	536.81	612.62
1500/ 680	**224/** 847	575.78	655.04
1600/ 725	**239/** 904	614.83	697.46
1700/ 771	**254/** 960	654.02	739.87
1800/ 816	**269/** 1017	693.31	782.25
1900/ 861	**284/** 1073	732.92	824.57
2000/ 907	**299/** 1130	772.77	866.87
2100/ 952	**313/** 1186	812.67	909.22
2200/ 997	**328/** 1243	852.78	951.62
2300/ 1043	**343/** 1299	893.07	994.09
2400/ 1088	**358/** 1356	933.48	1036.56
2500/ 1134	**373/** 1412	973.92	1079.03
2600/ 1179	**388/** 1469	1014.40	1121.49
2700/ 1224	**403/** 1525	1054.90	1163.96
2800/ 1270	**418/** 1582	1095.37	1206.43
2898/ 1314	**433/** 1637	1135.15	1248.00
2900/ 1315	**433/** 1638		1248.85

Figure 19-9. Usable Fuel Moments—Wing and Fuselage Tank (Sheet 1 of 2)

basic empty weight shall be corrected for these items. All items should be in their normal place during weighing.

6. The aircraft must be in a level attitude at the time of weighing. Level is determined with a plumb bob attached to a clip on the upper side of one of the frames in the tailcone and the level plate hole on the lower portion of the frame.

7. Weighing should always be made in an enclosed area free of drafts. The scales used should be properly certified and calibrated.

8. Weighing the aircraft on jacks.

    a. Three jackpoints are provided for weighing: Two on the wing at fuselage station 414.83, and one on the forward fuselage at fuselage station 101.50.

    b. Leveling is accomplished by adjusting the jacks.

9. Weighing the aircraft on wheels.

    a. Inflate the tires to the proper pressure.

    b. Deflate the shock struts. This is to establish the fuselage station of the centerline of the nosewheel axle. The fuselage station for the nosewheel axle in this condition is 92.72.

    c. Leveling is accomplished by inflating the main gear struts.

    d. Measure the perpendicular distance between the nose and main gear axle center lines. Add this measurement to the nose gear axle position (fuselage station 92.72) to obtain the main gear fuselage station.

    e. Substitute the nose and main gear stations for the jackpoint stations on the Aircraft Weighing Record and proceed as in jackpoint weighing.

POUNDS/KG	GALLONS/LITERS KEROSENE 6.7 LB/GAL	MOMENT/1,000 WING KEROSENE	MOMENT/1,000 FUSELAGE KEROSENE
3000/ 1360	448/ 1695		1291.20
3100/ 1406	463/ 1751		1333.43
3200/ 1451	478/ 1808		1375.68
3300/ 1496	493/ 1864		1418.05
3400/ 1542	507/ 1921		1460.45
3500/ 1587	522/ 1977		1502.93
3600/ 1633	537/ 2034		1545.30
3700/ 1678	552/ 2090		1587.66
3800/ 1723	567/ 2147		1630.07
3900/ 1769	582/ 2203		1672.56
4000/ 1814	597/ 2260		1715.00
4100/ 1859	612/ 2316		1757.29
4200/ 1905	627/ 2373		1799.51
4300/ 1950	642/ 2429		1841.70
4400/ 1995	657/ 2486		1883.90
4500/ 2041	672/ 2542		1925.98
4600/ 2086	687/ 2599		1967.94
4700/ 2131	701/ 2655		2009.80
4800/ 2177	716/ 2712		2051.89
4900/ 2222	731/ 2768		2093.69
5000/ 2268	746/ 2825		2135.36
5012/ 2273	748/ 2831		2140.42

Figure 19-9. Usable Fuel Moments—Wing and Fuselage Tank (Sheet 2 of 2)

UNUSABLE AND TRAPPED FLUIDS

The basic empty weight includes full oil, trapped, sump, and unusable fuel. If the aircraft is weighed after draining the fuel and oil, the sump and unusable fuel and drainable oil must be added to the "as weighed" condition to obtain the basic empty weight (Figure 19-10).

When the aircraft is weighed with completely dry fuel and drained oil systems, the trapped, sump, and unusable fuel and drainable oil weight and moment must be added to the "as weighed" condition to obtain the basic empty weight of the aircraft. This dry fuel condition would occur only if the fuel system were purged.

An Aircraft Weighing Record form is shown in Figure 19-11.

NOTES

UNUSABLE AND TRAPPED FUEL

ITEM	POUNDS	MOMENT/1,000
	KEROSENE 6.7 LB/GAL	KEROSENE
TRAPPED FUEL	12.1	4.44
DRAINABLE SUMP	11.9	4.61
INFLIGHT UNUSABLE	32.0	12.24
TOTAL	56.0	21.29

TRAPPED AND DRAINABLE ENGINE OIL

ITEM	GALLON	WEIGHT (8.0 LB/GAL)	MOMENT/1,000
TRAPPED OIL	0.5	4.0	1.9
DRAINABLE OIL	3.7	30.0	14.0
TOTAL OIL	4.2	34.0	15.9

Figure 19-10. Unusable and Trapped Fluids

DATE WEIGHED	MODEL	SERIAL NO.
PLACE WEIGHED	WEIGHING INSPECTOR	

REACTION	SCALE READING	TARE	NET WEIGHT	ARM	MOMENT
LEFT JACK					
RIGHT JACK					
SUBTOTAL				414.83*	
NOSE JACK				101.50*	
TOTAL ITEMS (AS WEIGHED)					
TOTAL ITEMS TABLE 1			−		
TOTAL ITEMS TABLE 2			+		
BASIC AIRPLANE					

TABLE 1

ITEMS WEIGHED BUT NOT PART OF BASIC PLANE	WEIGHT	ARM	MOMENT
JACK PADS			
TOTAL			

TABLE 2

BASIC ITEMS NOT IN WHEN WEIGHED	WEIGHT	ARM	MOMENT
USUABLE FUEL			
TOTAL			

* JACKPOINTS STATIONS FOR ELECTRONIC SCALES WEIGHING

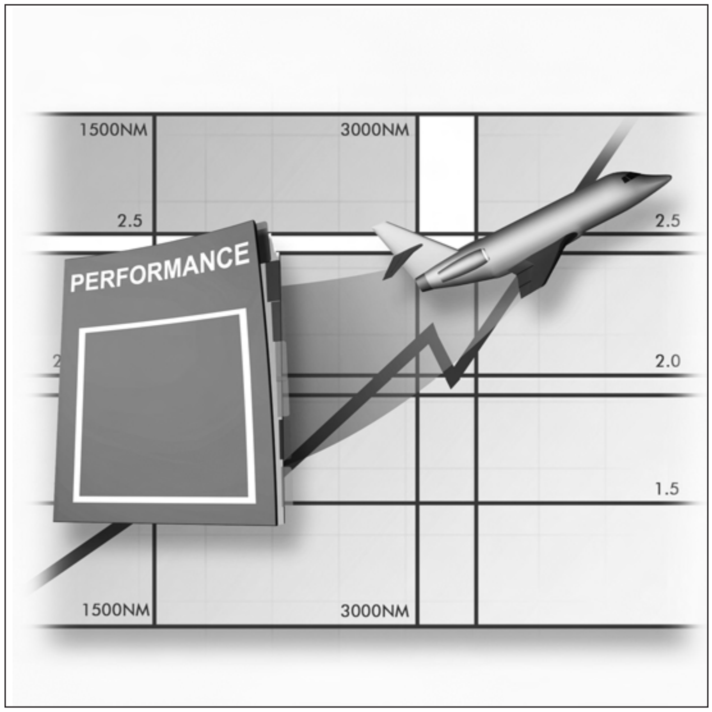
Figure 19-11. Aircraft Weighing Record Form

CHAPTER 20
PERFORMANCE

CONTENTS

TABLES

CHAPTER 20
PERFORMANCE



INTRODUCTION TO PERFORMANCE DATA

REGULATORY COMPLIANCE

Information in this chapter is presented for the purpose of compliance with the appropriate performance criteria and certification requirement of FAR 25.

STANDARD PERFORMANCE CONDITIONS

All performance in this chapter is based on flight test data and the following performance conditions:

1. Pertinent thrust ratings less installation, airbleed, and accessory losses.
2. Full temperature and altitude accountability within the operational limits for which the airplane is certified.

NOTE

Should OAT or altitude be below the lowest value shown on the performance charts, use performance at the lowest value shown.

3. Wing flap positions are as follows:

   Takeoff..8° or 20°

   Enroute .... UP 0°

   Approach .... 8° or 20°

   Landing....................................... DN 40°

4. Power settings (N_1) from the appropriate charts or tables and indicated by the N_1 bugs.

5. All takeoff and landing performance is based on a paved, dry runway.

6. The takeoff performance was obtained using the following procedures and conditions.

   Engine-Out Takeoff—Accelerate Go:

   a. The thrust lever was set static to the takeoff (T/O) power position, and then the brakes were released.

   b. The pilot recognized engine failure at V_1.

   c. The airplane continued to accelerate to V_R at which time positive rotation to +9° nose-up pitch attitude was made.

   d. The landing gear was retracted when a positive climb rate was established.

   e. V_2 was achieved at or prior to the 35-foot point above the runway.

   Engine-Out Takeoff—Accelerate Stop:

   a. The thrust lever was set static to the takeoff (T/O) power position, and then the brakes were released.

   b. The pilot recognized the necessity to stop because of engine failure just prior to V_1.

   c. Maximum pilot braking effort was started at V_1 and continued until the airplane came to a stop.

   d. Both thrust levers were brought to IDLE.

   e. The spoilers were deployed (either manual or auto).

   Multi-Engine Takeoff:

   a. The thrust levers were set static to the takeoff (T/O) power position, and then the brakes were released.

   b. Positive rotation to +9° nose-up pitch attitude was made at V_R.

   c. The landing gear was retracted when a positive climb rate was established.

   d. V_2 +8 KIAS was achieved at or prior to the 35-foot point above the runway.

7. The landing performance was obtained using the following procedures and conditions.

   Landing:

   a. Approached at V_{REF} with flaps and gear down using thrust as required to maintain a 3° glideslope. Targeted touchdown approximately 1,000 feet from end of runway.

   b. At the 50-foot point, a slight reduction in thrust was made. At 25 feet, thrust levers were briskly moved to IDLE.

   c. A firm touchdown was accomplished with little or no flare.

   d. After touchdown, maximum braking was applied until the airplane came to a full stop.

e. • *On aircraft 60-001 through 60-093, except 60-079, not incorporating SB 60-27-6*, spoilers were deployed manually.

• *On aircraft 60-079, 60-094 and subsequent, and prior aircraft incorporating SB 60-27-6*, spoilers were deployed manually or automatically as indicated on the applicable chart.

VARIABLE FACTORS AFFECTING PERFORMANCE

Details of variables affecting performance are given with the charts to which they apply. Conditions which relate to all performance calculations are:

1. Cabin air .... ON
2. Effect of humidity.
3. Winds, for which graphical correction is presented on the charts, are to be taken as the reported winds. Factors for 50% headwind component and 150% tailwind component have been applied as prescribed in pertinent regulations.
4. The percentages of stall speed are calculated from speeds as expressed in calibrated airspeed (KCAS).

DEFINITIONS

These definitions apply to terms used throughout this manual.

AIRSPEEDS

CAS — ***Calibrated Airspeed***—The airspeed indicator reading corrected for instrument and position error. KCAS is calibrated airspeed expressed in knots.

IAS — ***Indicated Airspeed***—The airspeed indicator reading as installed in the airplane. KIAS is indicated airspeed expressed in knots. The information in this manual is presented in terms of indicated airspeed, unless otherwise stated, and assumes zero instrument error. The ground airspeed calibration was adjusted to account for the lag in the electronic airspeed indicator.

M — ***Calibrated Mach Number***—The machmeter reading corrected for instrument and position error.

M_I — ***Indicated Mach Number***—The machmeter reading as installed in the airplane. Zero instrument error is assumed for presentations in this manual.

V_A — ***Maneuvering Speed***—V_A is the highest speed that full aileron and rudder control can be applied without overstressing the aircraft, or the speed at which the aircraft will stall with a load factor of 3.0g's, whichever is less.

V_{FE} — ***Maximum Flap Extended Speed***—The maximum speed permissible with the wing flaps in a prescribed extended position.

V_{LE} — ***Maximum Landing Gear Extended Speed***—The maximum speed at which the aircraft can be safely flown with the landing gear extended.

V_{LO} — ***Maximum Landing Gear Operating Speed***—The maximum speed at which the landing gear can be safely extended or retracted.

V_{LOF} — ***Lift-off Speed***—The actual speed of the aircraft at lift-off.

V_{MO}/M_{MO} ***Maximum Operating Limit Speed***—The speed that may not be deliberately exceeded in any regiment flight (climb, criuse, or descent) unless a higher speed is authorized for the flight test. V_{MO} is expressed in knots. M_{MO} is expressed in mach number.

V_{S0} ***V_{S0}*** is the stalling speed in the landing configuration.

V_{S1} ***V_{S1}*** is the stalling speed in the appropriate gear/flap configuration.

V_{MCA} ***Minimum Control Speed, Air***—The minimum flight speed at which the airplane is controllable with up to 5° of bank when one engine suddenly becomes inoperative and the remaining engine is operating at takeoff thrust.

V_{MCG} ***Minimum Control Speed, Ground***—The minimum speed on the ground at which control can be maintained using aerodynamic controls alone, when one engine suddenly becomes inoperative and the remaining engine is operating at takeoff thrust.

V_{MCL} ***Minimum Control Speed, Landing***—The minimum flight speed during landing approach at which the airplane is controllable with up to 5° of bank when one engine suddenly becomes inoperative and the remaining engine is operating at takeoff thrust V_{MCL} has been determined to be not more than 107 KIAS in the landing configuration.

V_1 ***Takeoff Decision Speed***—The speed at which the distance to continue the takeoff to 35 feet or the distance to stop will not exceed the scheduled takeoff distance provided the brakes are applied at V_1.

V_R ***Rotation Speed***—The speed at which rotation is initiated during takeoff performance.

V_2 ***Takeoff Safety Speed***—The actual speed at 35 feet above the runway surface as demonstrated in flight during single-engine takeoff. V_2 must not be less than 1.2 times the stalling speed, or less than 1.1 times the air minimum control speed (V_{MCA}), or less than the rotation speed (V_R) plus an increment in speed attained prior to reaching a 35-foot height above the runway surface.

V_{APP} ***Approach Climb Speed***—The airspeed equal to 1.3 V_{S1} (airplane in the approach configuration).

V_{REF} ***Landing Approach Speed***—The airspeed equal to 1.3 V_{S0} (airplane in the approach configuration).

1.3 V_S ***Landing Approach Speed*** for abnormal operations in which flaps are not full down. Values are presented for flaps 0°, 8°, and 20°. These speeds provide the same margin above stall as when flying a normal approach with flaps full down at V_{REF}.

WEIGHTS

Maximum Allowable Takeoff Weight—The maximum allowable takeoff weight at the start of takeoff roll is limited by the most restrictive of the following requirements:

- Maximum Certified Takeoff Weight
- Maximum Takeoff Weight (Climb or Brake Energy Limited) for altitude and reported surface temperature as determined from the applicable figure entitled Takeoff Weight Limits in this chapter
- Maximum Takeoff Weight for the runway and ambient conditions as determined from the applicable figure entitled Takeoff Field Length in this chapter

Maximum Allowable Landing Weight—The maximum allowable landing weight is limited by the most restrictive of the following requirements:

- Maximum Certified Landing Weight
- Maximum Landing Weight for the runway and ambient conditions as determined from the applicable Actual Landing Distance chart in this chapter
- Maximum Landing Weight (Approach Climb or Brake Energy Limited) for altitude and reported surface temperature as determined from the applicable figure entitled Landing Weight Limits in this chapter

DISTANCES

Accelerate-Stop Distance—The accelerate-stop distance is the horizontal distance traversed from brake release to the point at which the airplane comes to a complete stop on a takeoff during which the pilot applies the brakes at or below V_1.

Engine-Out Accelerate-Go Distance—The engine-out accelerate-go distance is the horizontal distance traversed from brake release to the point at which the airplane attains a height of 35 feet above the runway surface, on a takeoff during which one engine fails, recognition occurs at or above V_1 and the pilot elects to continue.

Takeoff Field Length—The takeoff field lengths presented on the Takeoff Field Length charts in this chapter are based on a smooth, dry, hard-surface runway. The takeoff field length given for each combination of airplane weight, atmospheric temperature, altitude, wind, and runway gradient is the greatest of the following:

1. 115% of the all-engine takeoff distance from start to a height of 35 feet above the runway surface
2. Accelerate-stop distance
3. Engine-out accelerate-go distance

No specific identification is made on the charts as to which of the above distances governs a specific case. However, in all cases for which charts are furnished, the field length is governed by either 2 or 3 above as 1 is shorter than either.

Generally, the takeoff field lengths presented are balanced; that is, the accelerate-go distance will be equal to the accelerate-stop distance. However, when V_1 is limited by either V_R or V_{MCG} the relationship is altered as follows:

- ***Accelerate-go Limited***—When V_1 is limited by V_R, the accelerate-go distance will be longer than the accelerate-stop distance.
- ***Accelerate-stop Limited***—When V_1 is limited by V_{MCG}, the accelerate-stop distance will be longer than the accelerate-go distance.

Actual Landing Distance—The actual landing distances presented in this chapter are based on a smooth, dry, hard-surface runway. The actual landing distance is equal to the horizontal distance from a point 50 feet above the runway surface to the point at which the airplane would come to a full stop on the runway.

Factored Landing Distance—The factored landing distances presented in this chapter are equal to the actual landing distance divided by 0.60 (multiplied by 1.67).

METEOROLOGICAL

ISA—International Standard Atmosphere.

OAT—Outside ambient air temperature obtained from ground meteorological sources. (OAT is equivalent to SAT.)

SAT (Static Air Temperature)—The inflight measured air temperature and includes full isentropic compression rise corresponding to the calibrated mach number. (SAT is equivalent to OAT.)

RAT (Ram Air Temperature)—The inflight measured air temperature and includes full isentropic compression rise corresponding to the calibrated mach number.

Altitude—All altitudes given in this chapter are pressure altitudes unless otherwise stated.

Wind—The wind velocities recorded as variables on the charts of this chapter are to be understood as the headwind or tailwind components of the actual winds at 10 meters above the runway surface (tower winds).

Demonstrated Crosswind—The demonstrated crosswind velocity of 29 knots is the velocity of the reported tower winds (measured at a 10-meter height) for which adequate control of the airplane during takeoff and landing (including the use of thrust reversers) was actually demonstrated during certification tests. Adequate control of the airplane during landing with spoilerons inoperative was demonstrated with a crosswind velocity of 25 knots. The values shown are not considered to be limiting.

MISCELLANEOUS

Position Correction (Static Position Correction)—A correction applied to indicated airspeed or altitude to eliminate the effect of the location of the static pressure source on the instrument reading. Any change in the airspeed-altitude system external to the airplane, or locating any external object near the pressure pickup sources, requires calibration of the system and revision of the charts.

Maximum Takeoff Brake Energy—The maximum brake energy demonstrated during emergency stop tests. Maximum effort braking at weights associated with the takeoff brake energies shown on the Takeoff Weight Limits charts will meet the accelerate-stop distances expressed on the Takeoff Field Length charts if the takeoff is aborted and brakes are applied at V_1. However, after the stop, wheel fuse plugs will release and brake and tire damage will occur.

Maximum Landing Brake Energy—The maximum brake energy demonstrated during landing tests. Maximum effort braking at weights associated with the landing brake energies shown on the Landing Weight Limits charts will meet the stopping distances expressed on the Actual Landing Distance chart. Landings in which brake energy is kept below this value can be accomplished without wheel fuse plug release or tire damage. Maximum effort stops in which brake energy exceeds this value may cause excessive brake wear, and after the stop, may cause wheel fuse plug release and damage tires.

Maximum Turn-Around Brake Energy—The maximum brake energy that when coupled with a rejected takeoff maintains wheel integrity. After maximum effort stops in which brake energy exceeds this value, as expressed on the applicable Landing Weight Limits chart, the brakes and wheels must be allowed to cool before attempting another takeoff. Refer to the turn-around limits presented in the “Limitations” chapter of this manual.

Runway Gradient—Change in runway elevation per 100 feet of runway length. The values given are positive for uphill gradients and negative for downhill gradients.

Gradient of Climb—The ratio of the change in height during a portion of the climb to the horizontal distance traversed in the same time interval.

Gross Climb Gradient—The climb gradient that the airplane can actually achieve given ideal conditions.

Net Climb Gradient—The gross climb gradient reduced by 0.8% during the takeoff phase and 1.1% enroute. This conservatism is required by FAR 25 for takeoff flight path determination to account for variables encountered in service.

First Segment Climb—Climb from the point at which the airplane becomes airborne to the point at which the landing gear is fully retracted. The gross climb gradient with one engine not operating and the other engine at takeoff thrust must be positive, without ground effect. This requirement is satisfied by observing the Takeoff Weight Limits chart. Airspeed increase is from liftoff airspeed (V_{LOF})

to V_2 with gradient calculated at the most limiting of the two conditions.

Second Segment Climb—Second segment climb is calculated at a height of 400 feet. The gross climb gradient may not be less than 2.4% with one engine not operating and the other engine at takeoff thrust. This requirement is satisfied by observing the Takeoff Weight Limits chart. Airspeed for this segment is V_2.

Final Segment Climb—Final segment climb is calculated at a height of 1,500 feet. The gross climb gradient may not be less than 1.2% with one engine not operating and the other engine at maximum continuous thrust. This requirement is satisfied by observing the Takeoff Weight Limits chart. Airspeed for this segment is 1.25 V_{S1}. The final segment climb gradients are used if the required obstacle clearance is greater than 1,500 feet AGL.

Enroute Climb—Climb with flaps UP (0°), landing gear retracted, and maximum continuous thrust on one engine. There is no minimum requirement for enroute climb gradients. The enroute net climb gradients are presented for pilot's reference. Airspeed is presented in the Enroute Climb Speed Schedule chart.

Approach Climb—Climb from a missed or aborted approach with approach (8°) flaps, landing gear retracted, and takeoff thrust on one engine. The gross climb gradient may not be less than 2.1%. This requirement is satisfied by observing the Landing Weight Limits chart. Airspeed for this segment is 1.3 V_{S1}.

Landing Climb—Climb from an aborted landing with landing flaps DN (40°), landing gear extended, and takeoff thrust on both engines. The gross climb gradient may not be less than 3.2%. This requirement is satisfied by observing the Landing Weight Limits chart. Airspeed for this segment is 1.3 V_{S0}.

The configurations referred to by name in the charts correspond to the settings listed in Table 20-1.

NOISE LEVELS

The noise levels are in compliance with the requirements of FAR 36 (Stage 3) which are essentially equivalent to the requirements outlines in ICAO Annex 16, Chapter 3.

No determination has been made by the Federal Aviation Administration that the noise levels in this manual are or should be acceptable or unacceptable for operation at, into, or out of any airport.

These noise values are stated for reference conditions of standard atmospheric pressure at sea level, 77°F (25°C) ambient temperature, 70% relative humidity, and zero wind.

Table 20-1. CONFIGURATION SETTINGS

CONFIGURATION	ENGINES OPERATING	THRUST	FLAP SETTING	GEAR
1st Segment Takeoff Climb	1	Takeoff	8° or 20°	DOWN
2nd Segment	1	Takeoff	8° or 20°	UP
Takeoff Climb				
Final Segment Climb	1	Max Cont	UP—0°	UP
Enroute Climb	1	Max Cont	UP—0°	UP
Approach Climb	1	Takeoff	8°	UP
Landing Climb	2	Takeoff	DN—40°	DOWN

CERTIFIED NOISE LEVELS

The noise levels established in compliance with FAR 36 (Stage 3) are listed in Table 20-2.

Table 20-2. CERTIFIED NOISE LEVEL IN EPNdB

CONDITION	ACTUAL	MAXIMUM ALLOWABLE
Sideline 23,100 lb (10,478 kg)	83.1	94
Takeoff 23,100 lb (10,478 kg)	70.8	89
Approach 19,500 lb (8845 kg)	87.7	98

Takeoff and sideline noise levels were obtained at the maximum takeoff weights listed above, 162 knots climb speed, 8° flaps, and anti-ice systems off. Thrust was reduced from takeoff N_1 %RPM to 79.2% N_1 at 2,500 feet AGL.

Landing approach noise levels were established with gear down, maximum landing weights listed above, approach speed of 140 knots, and 40° flaps. No special noise abatement procedures were used.

SUPPLEMENTAL NOISE LEVELS

The following noise levels provide additional information to the certified noise levels.

The noise levels established in compliance with FAR 36 (Stage 3) are listed in Table 20-3.

Table 20-3. SUPPLEMENTAL NOISE LEVEL IN EPNdB

CONDITION	ACTUAL	MAXIMUM ALLOWABLE
Sideline 23,500 lb (10,659 kg)	83.2	94
Takeoff 23,500 lb (10,659 kg)	78.9	89

Takeoff and sideline noise levels were obtained at the maximum takeoff weights listed above, 162 knots climb speed, 8° flaps, anti-ice systems off, and all engine takeoff with takeoff thrust setting.

CHAPTER 21
CREW RESOURCE MANAGEMENT (CRM)

CONTENTS

ILLUSTRATIONS

CHAPTER 21
CREW RESOURCE MANAGEMENT

CREW CONCEPT BRIEFING GUIDE

INTRODUCTION

Experience has shown that adherence to SOPs helps to enhance individual and crew cockpit situational awareness and will allow a higher performance level to be attained. Our objective is for standards to be agreed upon prior to flight and then adhered to, such that maximum crew performance is achieved. These procedures are not intended to supercede any individual company SOP, but rather are examples of good operating practices. See Figures 21-1 through 21-5.

PRETAKEOFF BRIEFING (IFR/VFR)

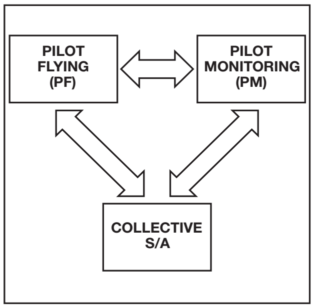
NOTE

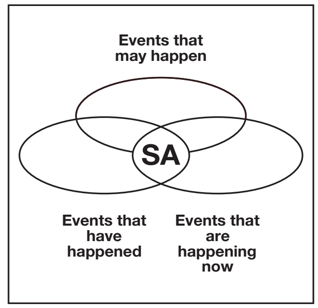
The following briefing is to be completed during item 1 of the Pretakeoff checklist. The pilot flying will accomplish the briefing.

1. Review the departure procedure (route and altitude, type of takeoff, significant terrain features, etc.).
2. Review anything out of the ordinary.
3. Review required callouts, unless standard calls have been agreed upon, in which case a request for "Standard Callouts" may be used.
4. Review the procedures to be used in case of an emergency on departure.
5. As a final item, ask if there are any questions.
6. State that the pretakeoff briefing is complete.

FLIGHTSAFETY CRM CARDS

SITUATIONAL AWARENESS IN THE COCKPIT





CLUES TO IDENTIFYING:

- Loss of Situational Awareness
- Links In the Error Chain

OPERATIONAL	1. FAILURE TO MEET TARGETS 2. UNDOCUMENTED PROCEDURE 3. DEPARTURE FROM SOP 4. VIOLATING MINIMUMS OR LIMITATIONS 5. FAILURE TO MONITOR
HUMAN	6. COMMUNICATIONS 7. AMBIGUITY 8. UNRESOLVED DISCREPANCIES 9. PREOCCUPATION OR DISTRACTION 10. CONFUSION OR EMPTY FEELING 11. NEED TO HURRY / LAST MINUTE CHANGES 12. FATIGUE

Figure 21-1. Situational Awareness in the Cockpit

COMMAND AND LEADERSHIP

LEADERSHIP STYLES VARY WITH SITUATION

AUTOCRATIC	AUTHORITARIAN	DEMOCRATIC	LAISSEZ-FAIRE

PARTICIPATION

LOW ⟷ **HIGH**

Command — Designated by Organization
— Cannot be Shared

Leadership — Shared among Crewmembers
— Focuses on "What's right," not "Who's right"

Figure 21-2. Command and Leadership

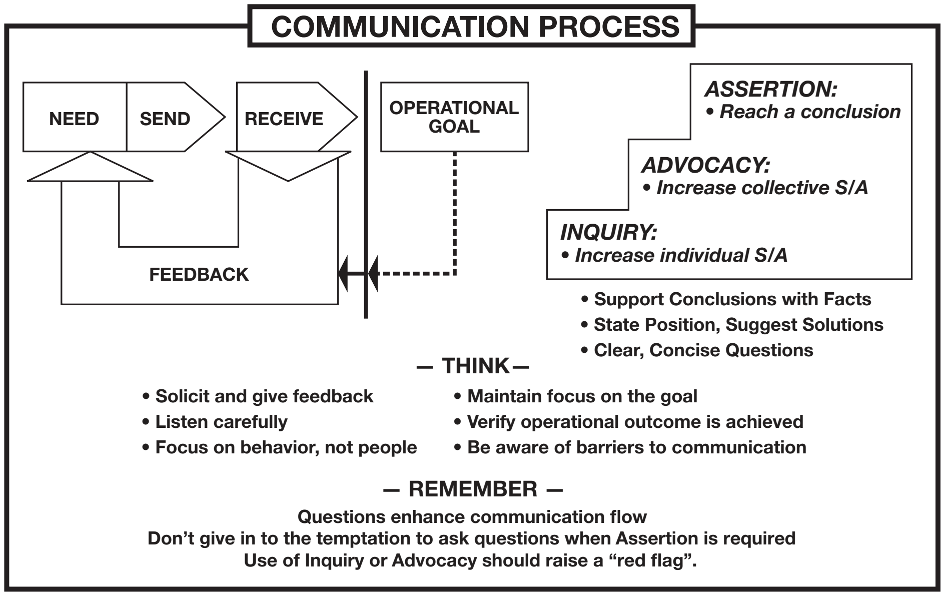


Figure 21-3. Communication Process

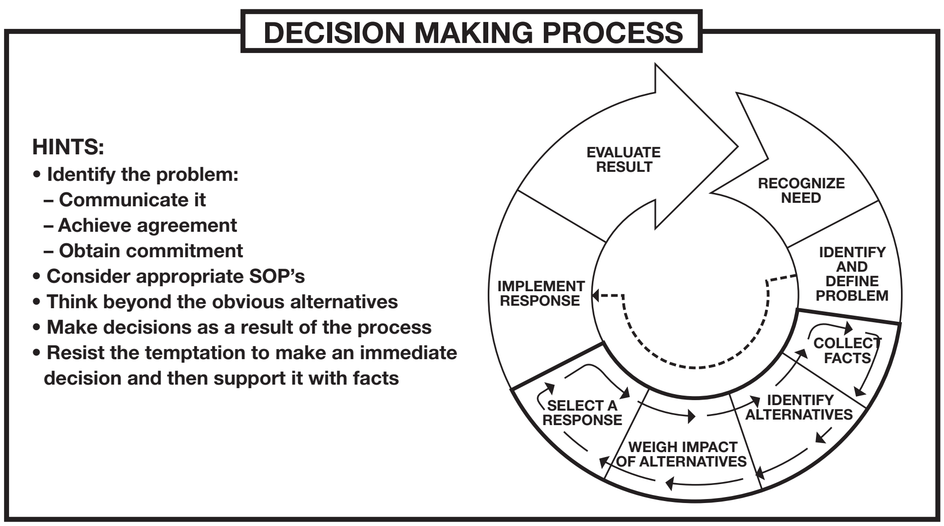


Figure 21-4. Decision Making Process

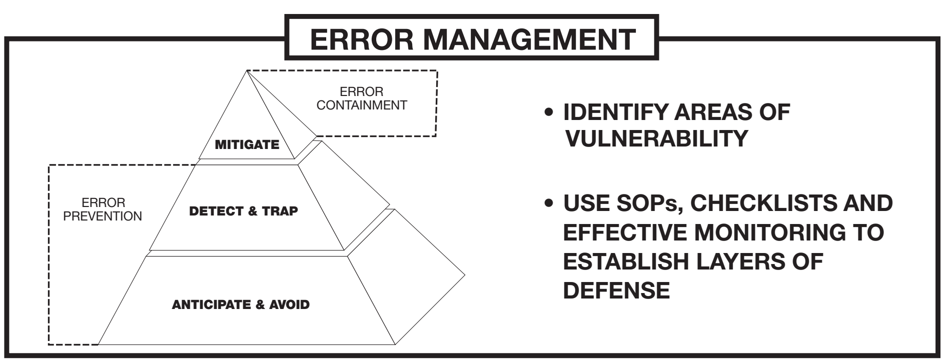


Figure 21-5. Error Management

CREW PERFORMANCE STANDARDS

SITUATIONAL AWARENESS

- Accomplishes appropriate pre-flight planning.
- Sets and monitors targets.
- Stays ahead of the aircraft by preparing for expected or contingency situations.
- Monitors weather, aircraft systems, instruments, and ATC communications.
- Shares relevant information with the rest of the crew.
- Uses advocacy/inquiry to maintain/regain situational awareness
- Recognizes error chain clues and takes action to break links in the chain.
- Communicates objectives and gains agreement when appropriate.
- Uses effective listening techniques to maintain/regain situational awareness.

STRESS

- Recognizes symptoms of stress in self and others.
- Maintains composure, calmness, and rational decision making under stress.
- Adaptable to stressful situations/personalities.
- Uses stress management techniques to reduce effects of stress.
- Maintains open, clear lines of communication when under stress.
- Manages low stress situations to prevent complacency and boredom.

COMMUNICATION

- Establishes open environment for interactive communications.
- Conducts adequate briefings to convey required information.
- Recognizes and works to overcome barriers to communications.
- Operational decisions are clearly stated to other crew memebers and acknowledged.
- Crewmembers are encouraged to state their own ideas, opinions and recommendations.
- Crewmembers are encouraged to ask questions regarding crew actions. Decisions and answers are provided openly and non-defensively.
- Assignment of blame is avoided. Focuses on **WHAT** is right, not **WHO** is right.
- Keeps feedback loop active until operational goal/decision is achieved.
- Conducts debriefings to correct substandard/inappropriate performance and to reinforce desired performance.

SYNERGY AND CREW CONCEPT

- Ensures that group climate is appropriate to operational situation.
- Coordinates flight crew activities to achieve optimum performance.
- Uses effective team building techniques.
- Demonstrates effective leadership and motivation techniques.
- Uses all available resources.
- Adapts leadership style to meet operational and human requirements.
- Encourages input/participation from all crewmembers.

WORKLOAD MANAGEMENT

- Communicates crew duties and receives acknowledgement.
- Sets priorities for crew activities.
- Recognizes and reports overloads in self and in others.
- Eliminates distractions in high workload situations.

- Maintains receptive attitude during high workload situations.
- Uses other crewmember.
- Avoids being a "one man show."

DECISION MAKING

- Anticipates problems in advance.
- Uses SOPs in decision making process.
- Seeks information from all available resources when appropriate.
- Avoids biasing source of information.
- Considers and weighs impact of alternatives.
- Selects appropriate courses of action in a timely manner.
- Evaluates outcome and adjusts/reprioritizes.
- Recognizes stress factors when making decisions and adjusts accordingly.
- Avoids making a decision and then going in search of facts that support it.

ADVANCED/AUTOMATED COCKPITS

- Follows automation related SOPs.
- Specifies pilot and copilot duties and responsibilities with regard to automation.
- Verbalizes and acknowledges entries and changes in flight operation.
- Verifies status and programming of automation.
- Selects appropriate levels of automation.
- Programs automation well in advance of maneuvers.
- Recognizes automation failure/invalid output indications.

BRIEFING GUIDE

LESSON REVIEW/ INTRODUCTION

- Previous Lesson Review
- Current Lesson Introduction
- Technical Objectives
- Human Factor Objectives
- Establish Realism

FLIGHT SCENARIO

- Flight Scenario Introduction
- Aircraft Data
- Flight Plan
- Previous Session Integrated
- New Elements Covered
- Flight Profile Items Summarized

TECHNICAL/NON-TECHNICAL INTEGRATION

- Technical Integration
- CRM Integration

CREW FLIGHT PLANNING

- Takeoff and Landing Data (TOLD) Card
- Flight Plan
- Fuel
- Crew Briefing

SUMMARY

- Main Briefing Points Review
- Commitment to Perform
- Summary Statement Delivered

DEBRIEFING GUIDE

DEBRIEFING AGENDA INTRODUCTION

- Time Limit Set
- Flight Profile Overview
- Topics for Debriefing Determined

SELF APPRAISAL/DISCOVERY

- Client Self Appraisal
- Client Self Discovery
- Client Performance vs. Standards

TECHNICAL/NON-TECHNICAL INTEGRATION TOOLS

- Mapping/Track Plot Used
- CMMS/FlightViz™/SimVu Used
- CRM Poster/CRM Concept Card Used
- CRM Performance Standards Card Used

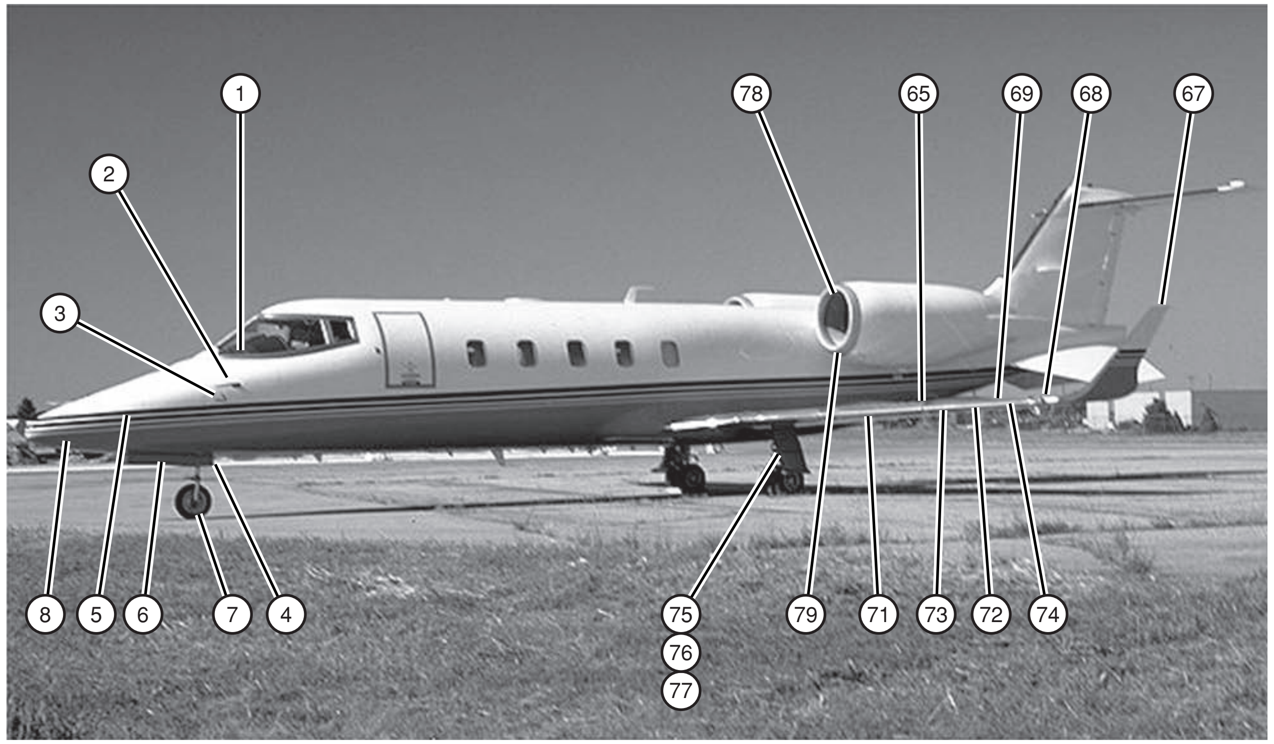
DEBRIEFING CONCLUSION

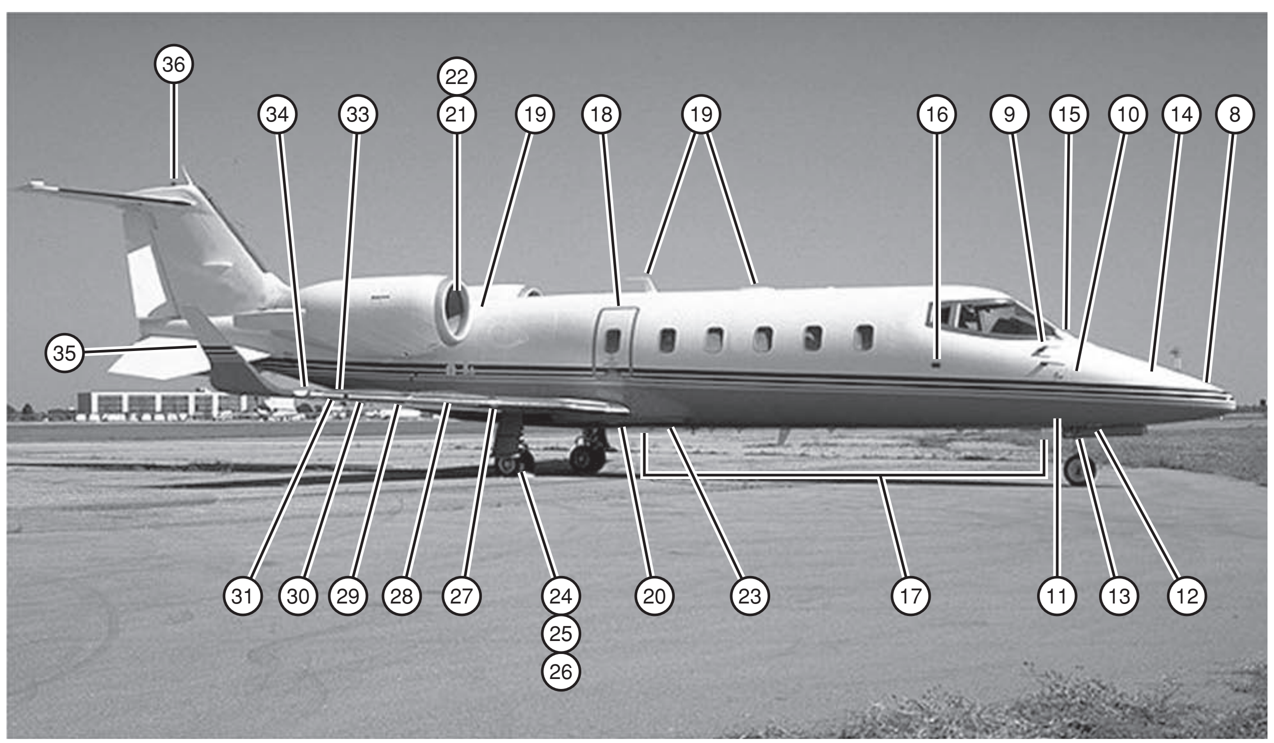
- Debriefing Points Review
- Next Event Preview
- Concluding Comments

WALKAROUND

The following section is a pictorial walkaround. It shows each item called out in the exterior power-off preflight inspection. The fold-out pages at the beginning and the end of the walkaround section should be unfolded before starting to read.

The general location photographs do not specify every checklist item. However, each item is portrayed on the large-scale photographs that follow.





WALKAROUND INSPECTION

2

1. **PILOT'S WINDSHIELD ALCOHOL DISCHARGE OUTLETS AND PILOT'S ANTI-ICE OUTLET —** CLEAR OF OBSTRUCTIONS.

2. **LEFT PITOT-STATIC PROBE —** COVER REMOVED, CLEAR OF OBSTRUCTIONS.
3. **LEFT STALL WARNING VANE —** FREEDOM OF MOVEMENT, LEAVE IN DOWN POSITION.

4. **LEFT PITOT-STATIC DRAIN VALVES (2) —** DRAIN.

4A. **OXYGEN SYSTEM DISCHARGE INDICATOR —** CONDITION.
5. **NOSE COMPARTMENT DOORS —** SECURE.

6. **NOSE GEAR AND WHEEL WELL —** HYDRAULIC LEAKAGE AND CONDITION, AND COOLING VENTS CLEAR.

7. **NOSE WHEEL AND TIRE —** CONDITION AND NOSE GEAR UPLOCK FORWARD.

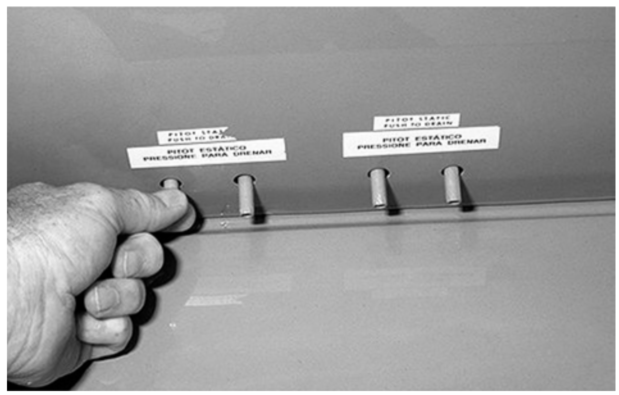
3

8. RADOME AND RADOME EROSION SHOE — CONDITION.

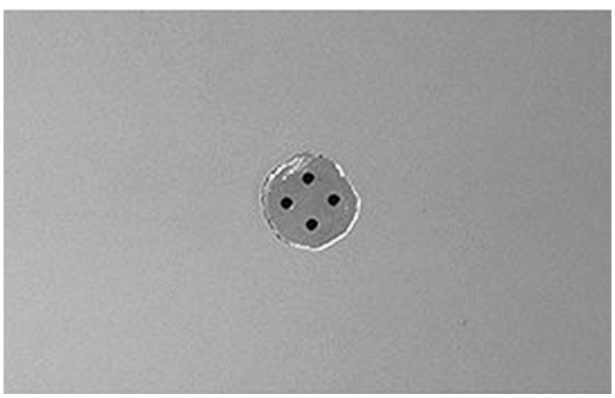
4

9. RIGHT PITOT-STATIC PROBES — COVERS REMOVED, CLEAR OF OBSTRUCTIONS.

10. RIGHT STALL WARNING VANE — FREEDOM OF MOVEMENT, LEAVE IN DOWN POSITION.



12. RIGHT PITOT-STATIC DRAIN VALVES (4) — DRAIN.



11. PRESSURIZATION STATIC PORT — CLEAR OF OBSTRUCTIONS.

13. TOTAL TEMPERATURE PROBE — CONDITION.

14. **NOSE COMPARTMENT DOORS —** SECURE.

16. **WING INSPECTION LIGHT AND LENS —** CONDITION.

15. **COPILOT'S WINDSHIELD ANTI-ICE OUTLET —** CLEAR OF OBSTRUCTIONS.

17. **LOWER FUSELAGE ANTENNAS, ROTATING BEACON LIGHT AND LENS —** CONDITION.

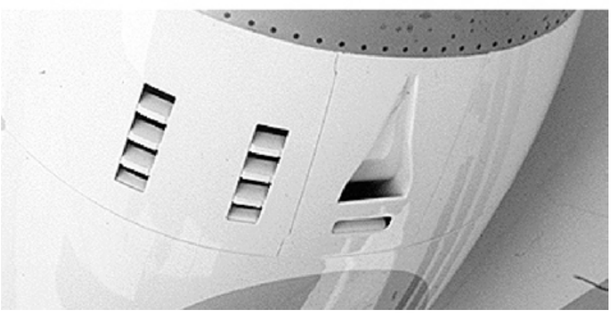
5

18. **AFT CABIN DOOR —** CONDITION.

19. **UPPER FUSELAGE ANTENNAS, AND DORSAL INLETS —** CONDITION.

19A. **FUSELAGE TANK ACCESS DOOR —** SECURE.

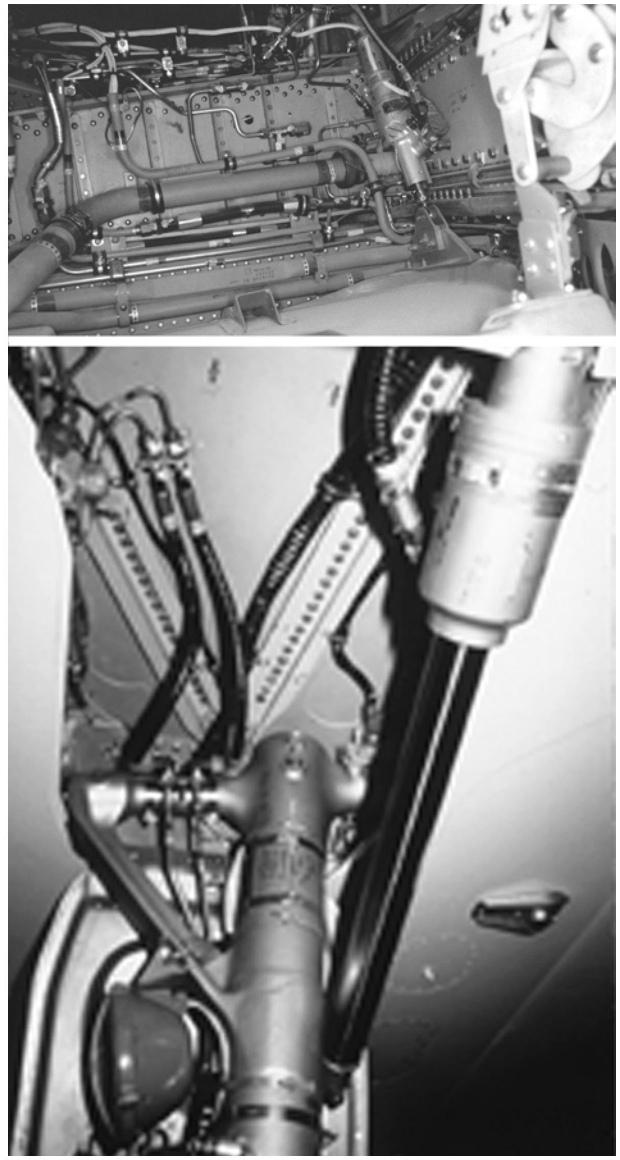
20. TOILET SERVICE DOOR — SECURE.



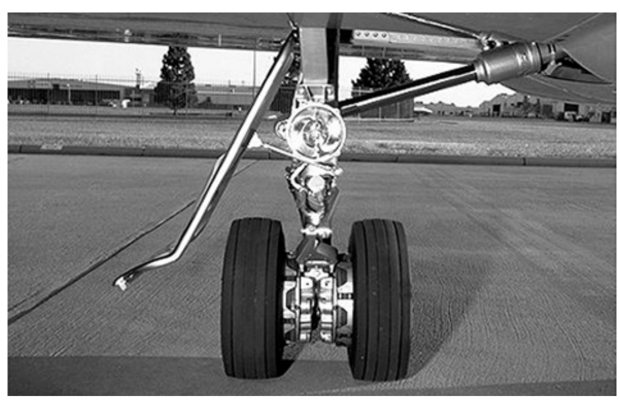
21. RIGHT ENGINE INLET AND FAN — CLEAR OF OBSTRUCTIONS AND CONDITION.

22. GENERATOR COOLING SCOOP — CLEAR

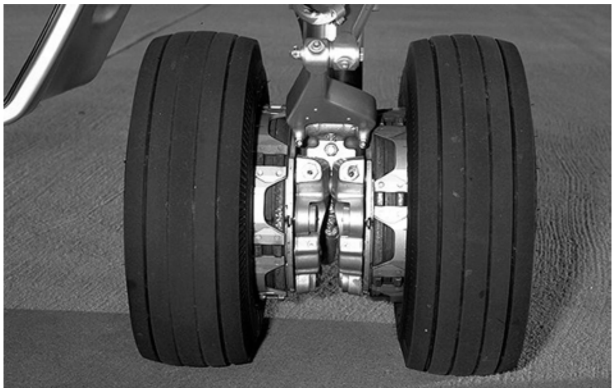
23. FUEL CROSSOVER DRAIN VALVE, WING SCAVENGE PUMP DRAIN VALVES (2), WING SUMP DRAIN VALVES (2), AND ENGINE FUEL DRAIN VALVES (2) — DRAIN.



24. RIGHT MAIN GEAR AND WHEEL WELL — HYDRAULIC/ FUEL LEAKAGE AND CONDITION.



25. RIGHT MAIN GEAR LANDING LIGHT AND DOORS — CONDITION

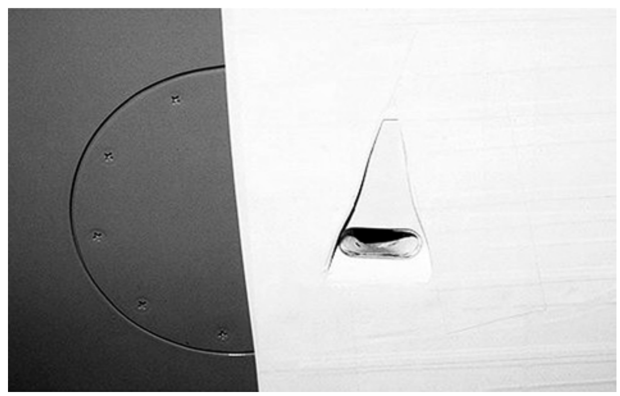


26. RIGHT MAIN GEAR WHEELS, BRAKES, AND TIRES — CONDITION.

6

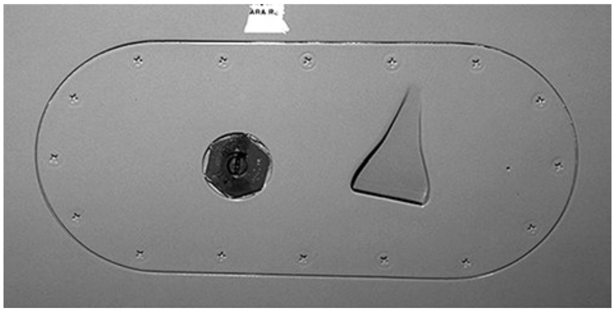
27. WING STALL FENCES — CONDITION.

28. LEADING-EDGE — CONDITION.

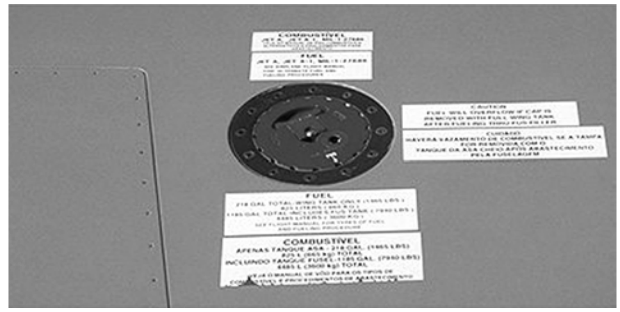


29. INBOARD FUEL VENT RAM AIRSCOOP — CLEAR OF OBSTRUCTIONS.

30. RIGHT WING ACCESS PANELS (UNDERSIDE OF WING) — CHECK FOR FUEL LEAKAGE.



31. OUTBOARD FUEL VENT RAM AIRSCOOP — CLEAR OF OBSTRUCTIONS. **OUTBOARD VENT SUMP —** DRAIN.



33. RIGHT WING FUEL FILLER CAP — CONDITION AND SECURITY.

32. INTENTIONALLY LEFT BLANK.

7

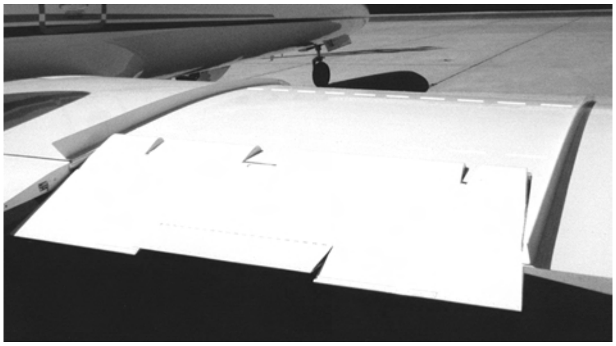
34. RIGHT WINGLET NAVIGATION LIGHT — CONDITION.

8

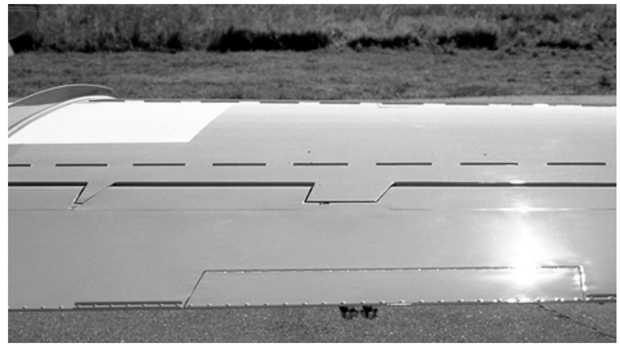
35. RIGHT WINGLET STATIC DISCHARGE WICKS (4) — CONDITION.

36. RECOGNITION AND BEACON/STROBE LIGHTS AND LENS — CONDITION.

9



37. RIGHT AILERON — CHECK FREE MOTION, BALANCE TAB LINKAGE, AND BRUSH SEAL CONDITION.

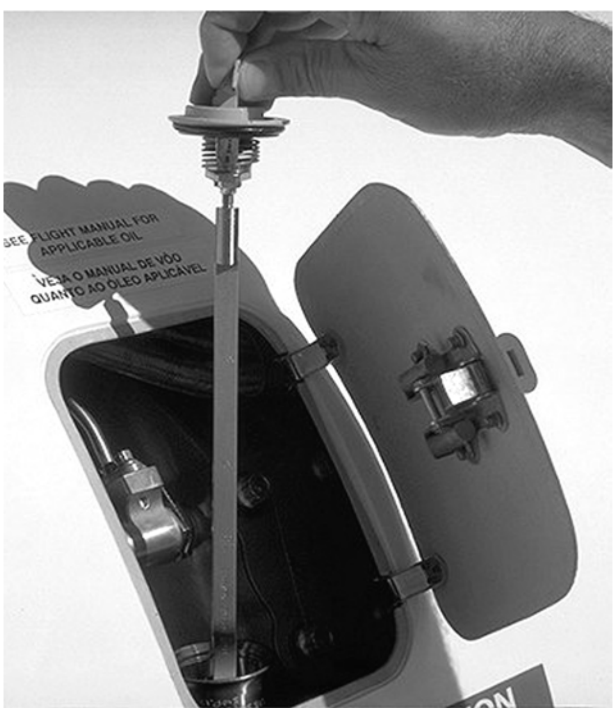


38. BOUNDARY LAYER ENERGIZERS — CONDITION.

10

39. RIGHT SPOILER AND FLAP — CONDITION.

40. RIGHT ENGINE OIL SERVICE ACCESS DOOR — OPEN ACCESS DOOR. CHECK OIL LEVEL (NORMAL) AND OIL TANK FILLER CAP SECURITY. SECURE ACCESS DOOR.



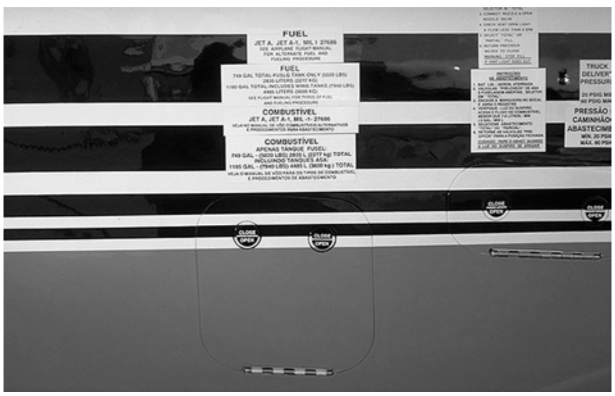
41. RIGHT ENGINE OIL — CHECK.

11

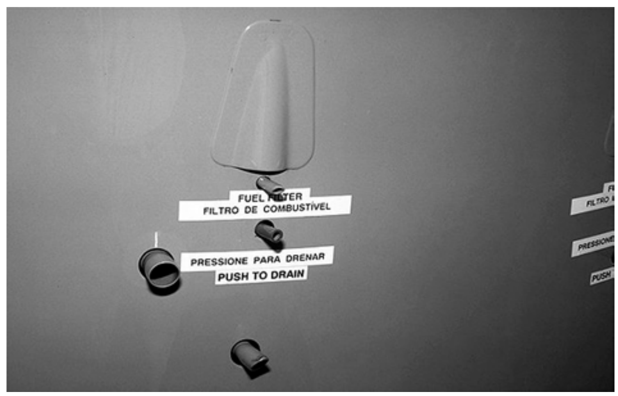
42. **RIGHT ENGINE TURBINE EXHAUST AREA —** CONDITION, CLEAR OF OBSTRUCTIONS. **RIGHT THRUST REVERSER —** CONDITION, COMPLETELY STOWED.

45. **FUSELAGE TANK SUMP DRAIN VALVE, EXPANSION LINE DRAIN VALVES (2) AND TRANSFER LINE DRAIN VALVES (2) —** DRAIN.

12

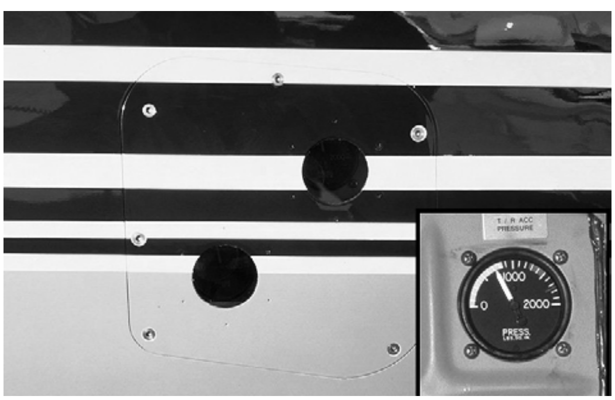


43. **SINGLE-POINT FUELING ACCESS DOORS (IF APPLICABLE) —** SECURE.



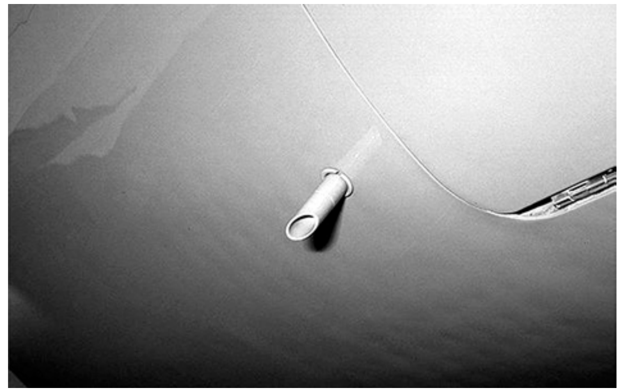
46. **FUEL FILTER DRAIN VALVES (2) —** DRAIN.

47. **FUSELAGE FUEL VENT —** CLEAR.



44. **HYDRAULIC SERVICE ACCESS PANEL —** CHECK HYDRAULIC ACCUMULATOR PRESSURE (750 PSI MINIMUM).

48. HEAT EXCHANGER VENT SCREEN — CLEAR.



49. BATTERY VENTS — CLEAR.

13

50. RIGHT VOR/LOC ANTENNA — CONDITION. **RIGHT ELT ANTENNA (IF APPLICABLE) —** CONDITION.

52. STATIC DISCHARGE WICKS (6 ON ELEVATORS, 1 ABOVE NAV LIGHT, AND 4 ON DELTA FINS) — CONDITION.

53. VERTICAL FIN NAVIGATION LIGHT AND LENS — CONDITION.

14

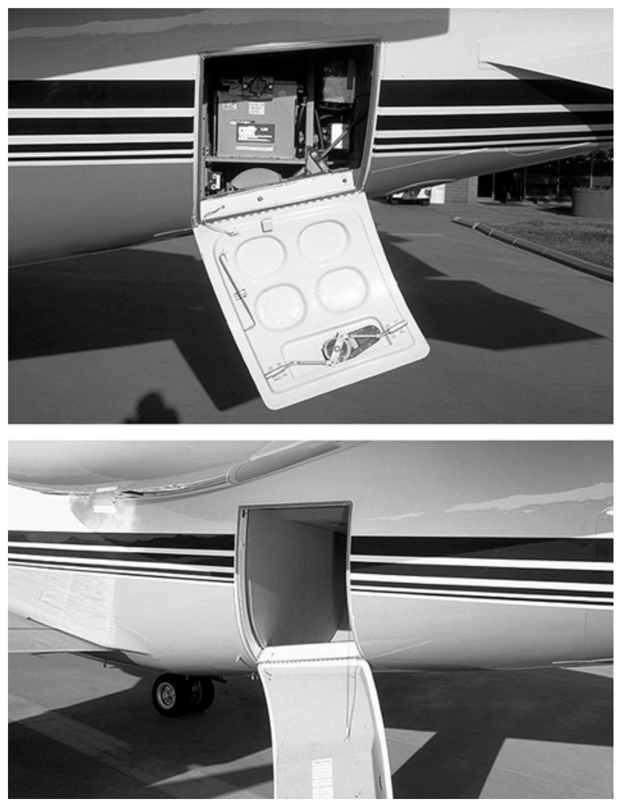
51. VERTICAL STABILIZER, RUDDER, HORIZONTAL STABILIZER, AND ELEVATOR — CONDITION, DRAIN HOLES CLEAR.

15

54. DELTA FINS — CONDITION.

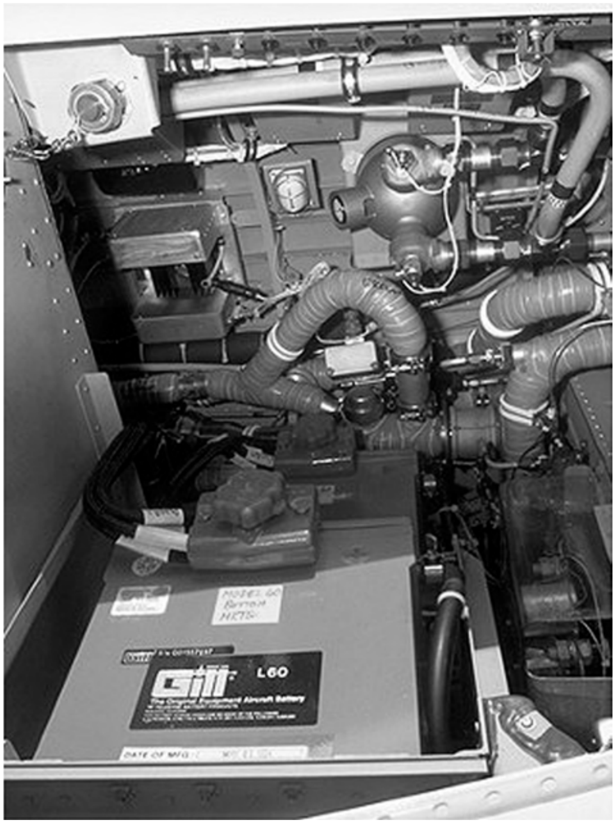
55. LEFT VOR/LOC ANTENNA — CONDITION.
56. HF ANTENNA — CONDITION.

54A. TAILSTAND — REMOVE.



57. TAILCONE ACCESS DOOR — OPEN.

WALKAROUND



58. **TAILCONE INTERIOR —** CHECK FOR FLUID LEAKS, SECURITY, AND CONDITION OF INSTALLED EQUIPMENT.

59. **TAILCONE ACCESS DOOR —** CLOSE AND SECURE.

60. **AFT BAGGAGE COMPARTMENT DOOR —** SECURE.

16

61. **ENGINE FIRE EXTINGUISHER DISCHARGE INDICATORS —** CONDITION.

17

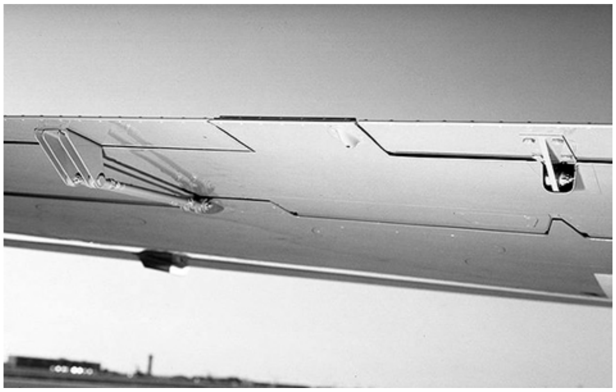
62. **LEFT ENGINE TURBINE EXHAUST AREA —** CONDITION, CLEAR OF OBSTRUCTIONS. **LEFT THRUST REVERSER —** CONDITION AND COMPLETELY STOWED.

18

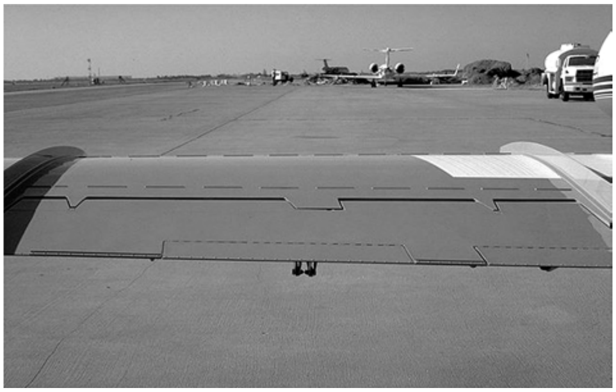
63. **LEFT ENGINE OIL SERVICE ACCESS DOOR —** OPEN ACCESS DOOR. CHECK OIL LEVEL (NORMAL). SECURE ACCESS DOOR.

64. **LEFT SPOILER AND FLAP —** CONDITION.

19



65. **LEFT AILERON —** CHECK FREE MOTION, BALANCE AND TRIM TAB LINKAGE, AND BRUSH SEAL CONDITION.

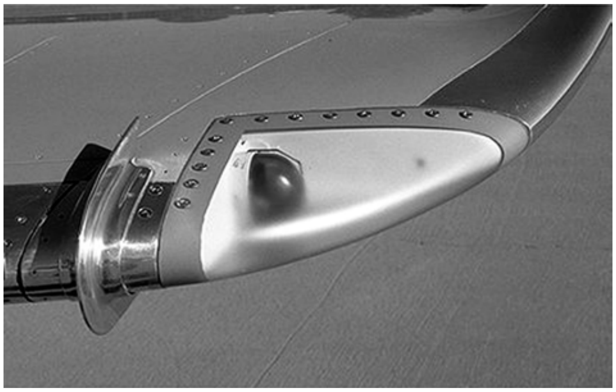


66. **BOUNDARY LAYER ENERGIZERS —** CONDITION.

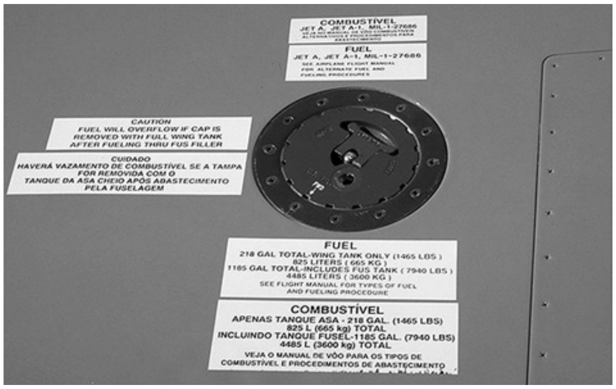
20

67. **LEFT WINGLET STATIC DISCHARGE WICKS (4) —** CONDITION.

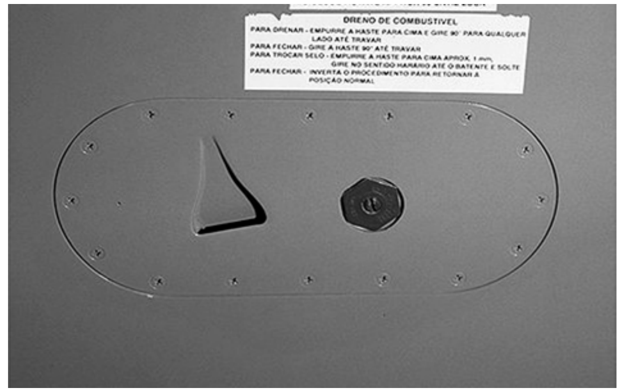
21



68. **LEFT WINGLET NAVIGATION LIGHT, STROBE LIGHT AND LENS —** CONDITION.



69. LEFT WING FUEL FILLER CAP — CONDITION AND SECURITY.



70. OUTBOARD FUEL VENT RAM AIRSCOOP — CLEAR OF OBSTRUCTIONS. **OUTBOARD VENT SUMP —** DRAIN.

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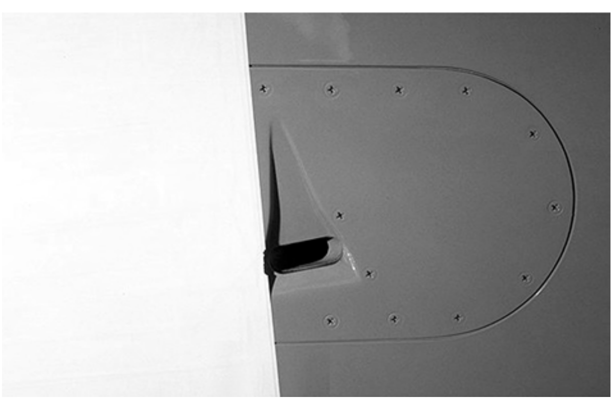
(22)

71. LEFT WING ACCESS PANELS — CHECK FOR FUEL LEAKAGE.

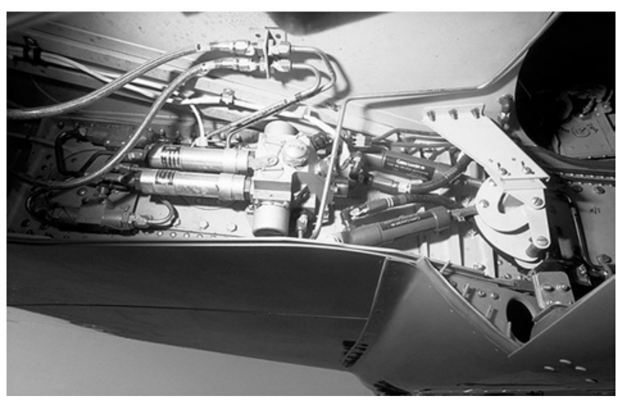
73. LEADING EDGE — CONDITION.



72. INBOARD FUEL VENT RAM AIRSCOOP — CLEAR OF OBSTRUCTIONS.

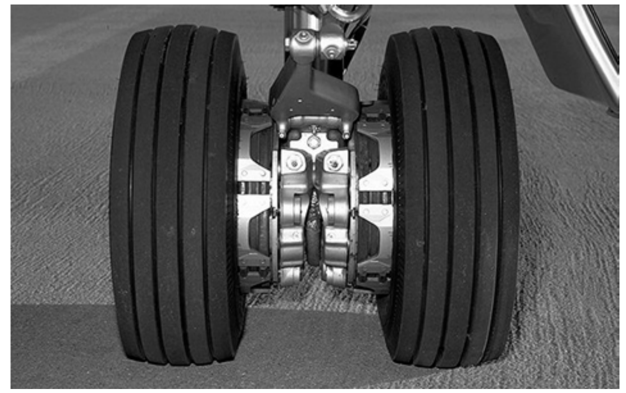
74. WING STALL FENCES — CONDITION.

75. LEFT MAIN GEAR AND WHEEL WELL — HYDRAULIC/FUEL LEAKAGE AND CONDITION.

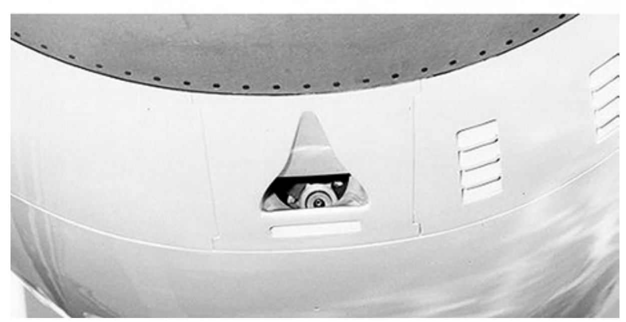


75A. WHEEL WELL HYDRAULIC — FUEL LEAKAGE AND CONDITION.

76. LEFT MAIN GEAR LANDING LIGHT AND DOORS — CONDITION.

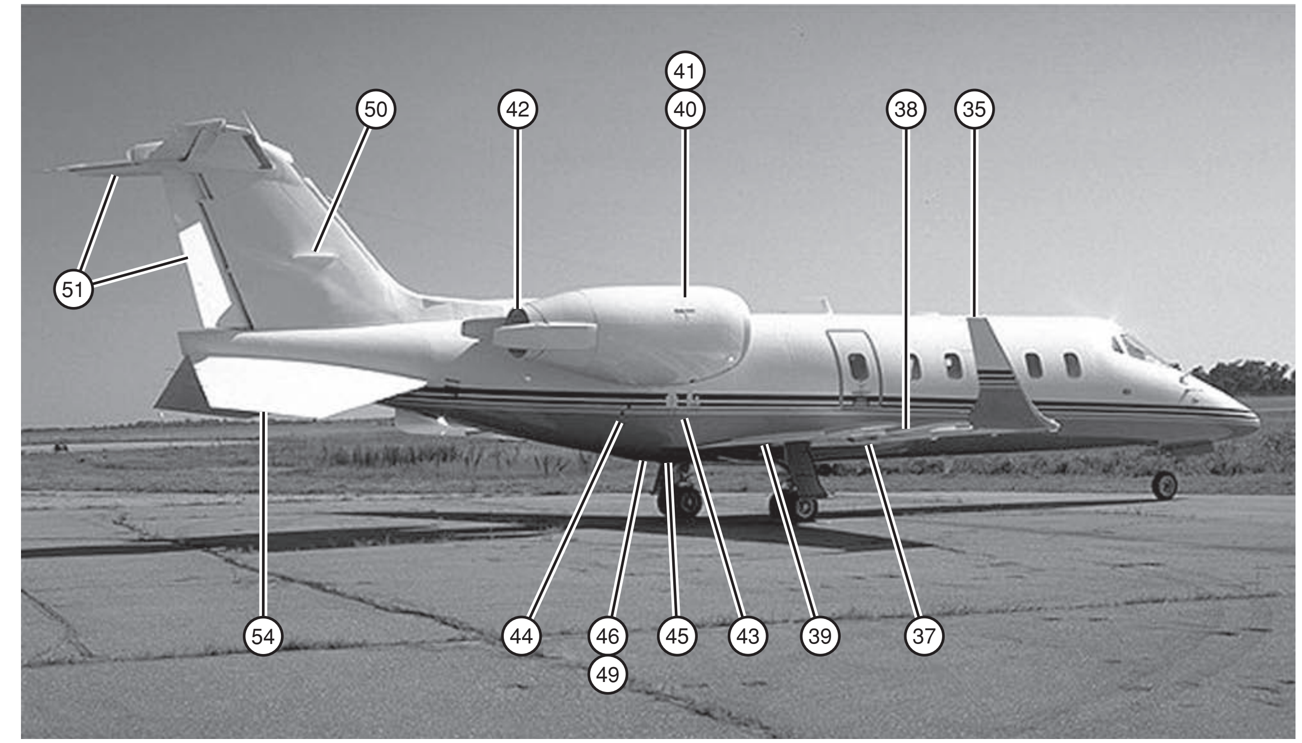


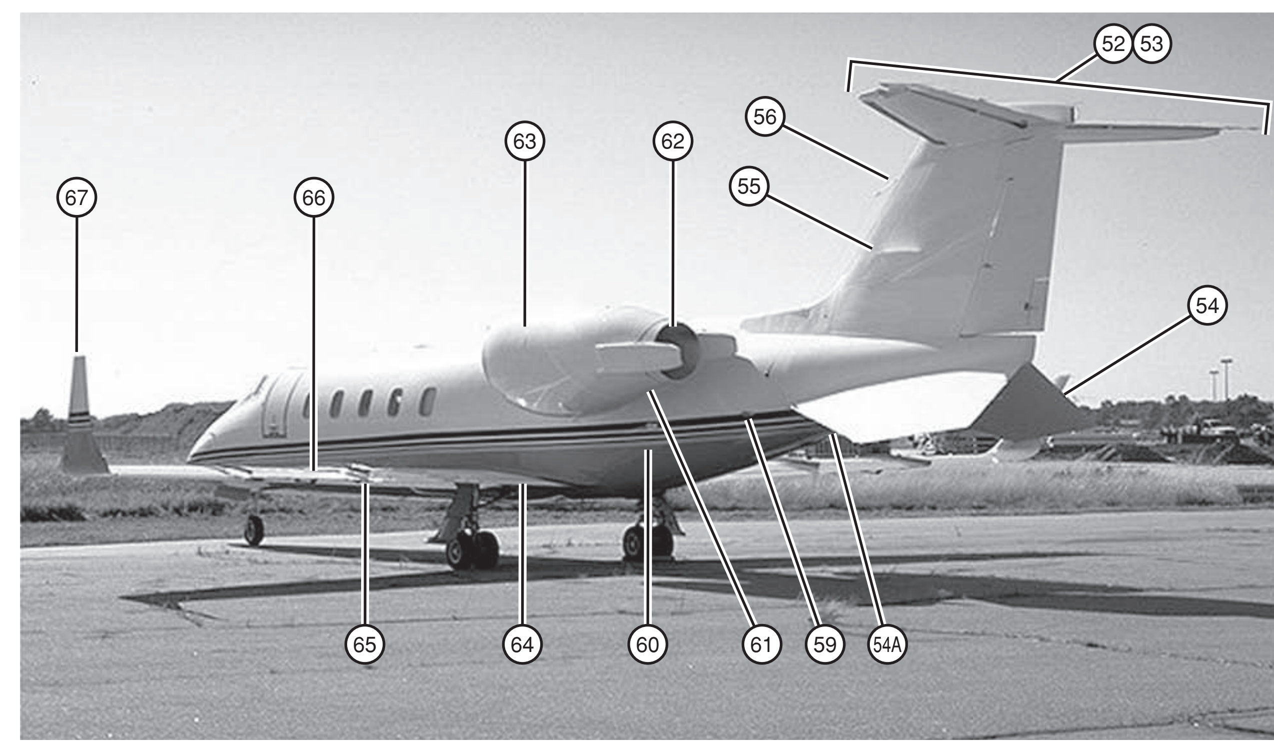
77. LEFT MAIN GEAR WHEELS, BRAKES, AND TIRES — CONDITION.



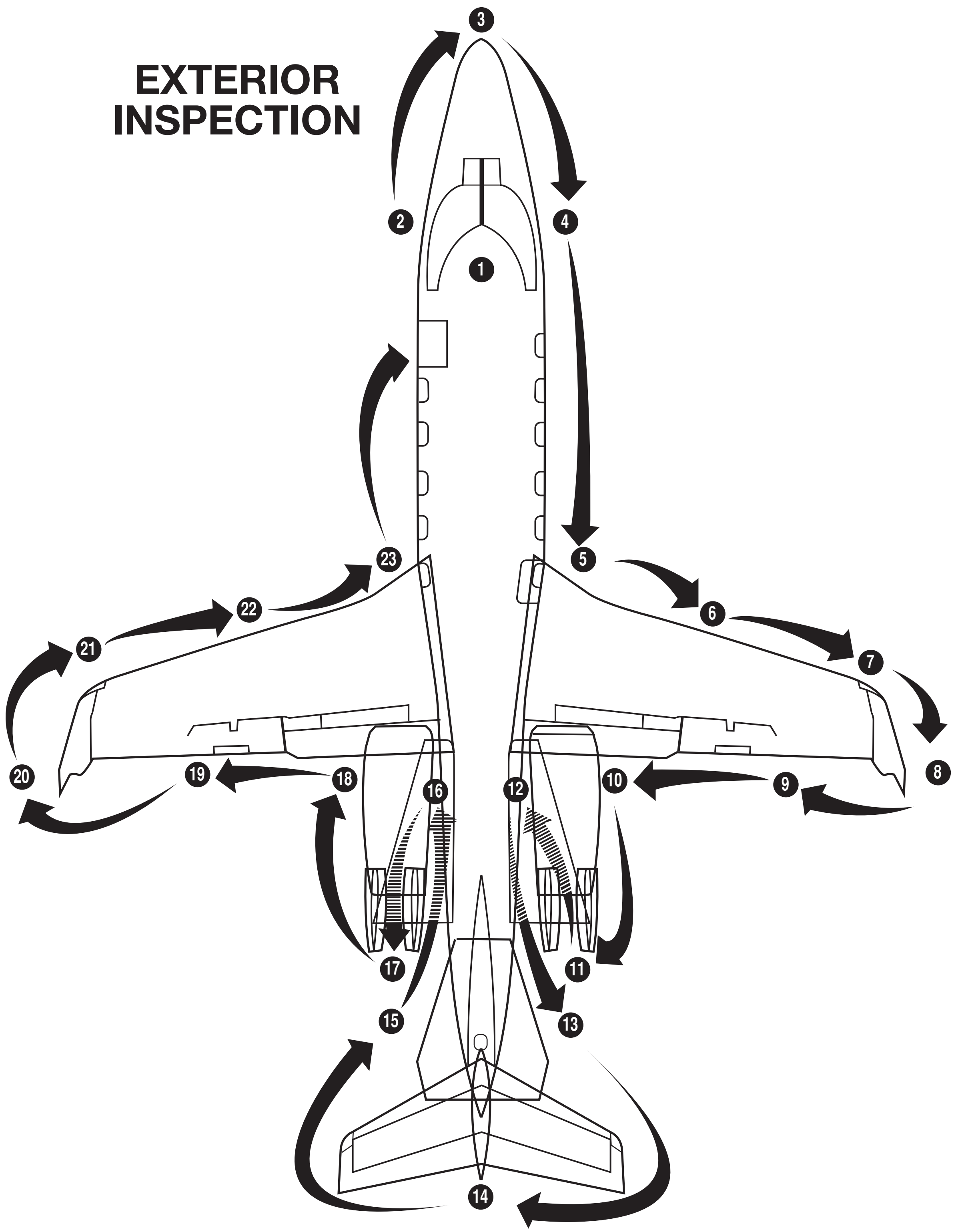
78. LEFT ENGINE INLET AND FAN — CLEAR OF OBSTRUCTIONS AND CONDITION.

79. GENERATOR COOLING SCOOP — CLEAR.





EXTERIOR INSPECTION



APPENDIX

CONTENTS

This appendix contains the following conversion tables:

Table APP-1. CONVERSION FACTORS

MULTIPLY	BY	TO OBTAIN
CENTIMETERS	0.3937	INCHES
KILOGRAMS	2.2046	POUNDS
KILOMETERS	0.621	STATUTE MILES
KILOMETERS	0.539	NAUTICAL MILES
LITERS	0.264	GALL0NS
LITERS	1.05	QUARTS (LIQUID)
METERS	39.37	INCHES
METERS	3.281	FEET
MILIBARS	0.02953	IN. HG (32°F)
FEET	0.3048	METERS
GALLONS	3.7853	LITERS
INCHES	2.54	CENTIMETERS
IN. HG (32°F)	33.8639	MILIBARS
NAUTICAL MILES	1.151	STATUTE MILES
NAUTICAL MILES	1.852	KILOMETERS
POUNDS	0.4536	KILOGRAMS
QUARTS (LIQUID)	0.946	LITERS
STATUTE MILES	1.609	KILOMETERS
STATUTE MILES	0.868	NAUTICAL MILES

INTENTIONALLY LEFT BLANK

ANSWERS TO QUESTIONS

CHAPTER 2

1. D
2. B
3. D
4. B
5. D
6. B
7. D
8. D
9. D
10. B
11. B
12. D
13. C
14. C
15. A

CHAPTER 3

1. C
2. B
3. C
4. D
5. B
6. B
7. C
8. D
9. D
10. D

CHAPTER 4

1. C
2. D
3. B
4. D
5. A

CHAPTER 5

1. B
2. A
3. C
4. C
5. D
6. B
7. D
8. B
9. D
10. D
11. A
12. D
13. C
14. D
15. B
16, D
17. D
18. A

CHAPTER 6

1. B
2. A
3. D
4. C
5. B
6. D
7. B
8. D

CHAPTER 7

1. D
2. C
3. B
4. D
5. D
6. A
7. C
8. D
9. B
10. D
11. A
12. D
13. D
14. C
15. D
16. D
17. C
18. D
19. B
20. D
21. A
22. D
23. A
24. B

CHAPTER 8

1. C
2. A
3. B
4. D

CHAPTER 9

1. D
2. D
3. A
4. D
5. A
6. D
7. A

CHAPTER 10

1. B
2. D
3. B
4. D
5. D
6. A
7. A
8. D

CHAPTER 11

1. A
2. D
3. B
4. B
5. A
6. A
7. D
8. A
9. D

CHAPTER 12

1. C
2. D
3. B
4. C
5. A
6. D
7. A
8. C

CHAPTER 13

1. B
2. A
3. D
4. A
5. C
6. B
7. A
8. C
9. D
10. D

CHAPTER 14

1. C
2. A
3. A
4. D
5. B
6. C
7. B
8. C
9. A
10. B
11. B
12. A
13. C
14. D
15. A
16. A

CHAPTER 15

1. D
2. B
3. C
4. C
5. A
6. D
7. D
8. C
9. B
10. D
11. C
12. D
13. C
14. C
15. D
16. D
17. C
18. D
19. C
20. D
21. D
22. A

CHAPTER 16

1. B
2. C
3. B
4. D
5. A
6. C
7. C
8. A
9. A
10. A
11. D
12. B
13. A
14. D
15. A
16. B
17. A
18. C or D
19. B
20. B

CHAPTER 17

1. B
2. A
3. B
4. C
5. B

ANNUNCIATORS

The Annunciator Section presents a color representation of all the annunciator lights in the aircraft.

Please unfold page ANN-1 to the right and leave it open for ready reference as the annunciators are cited in the text.

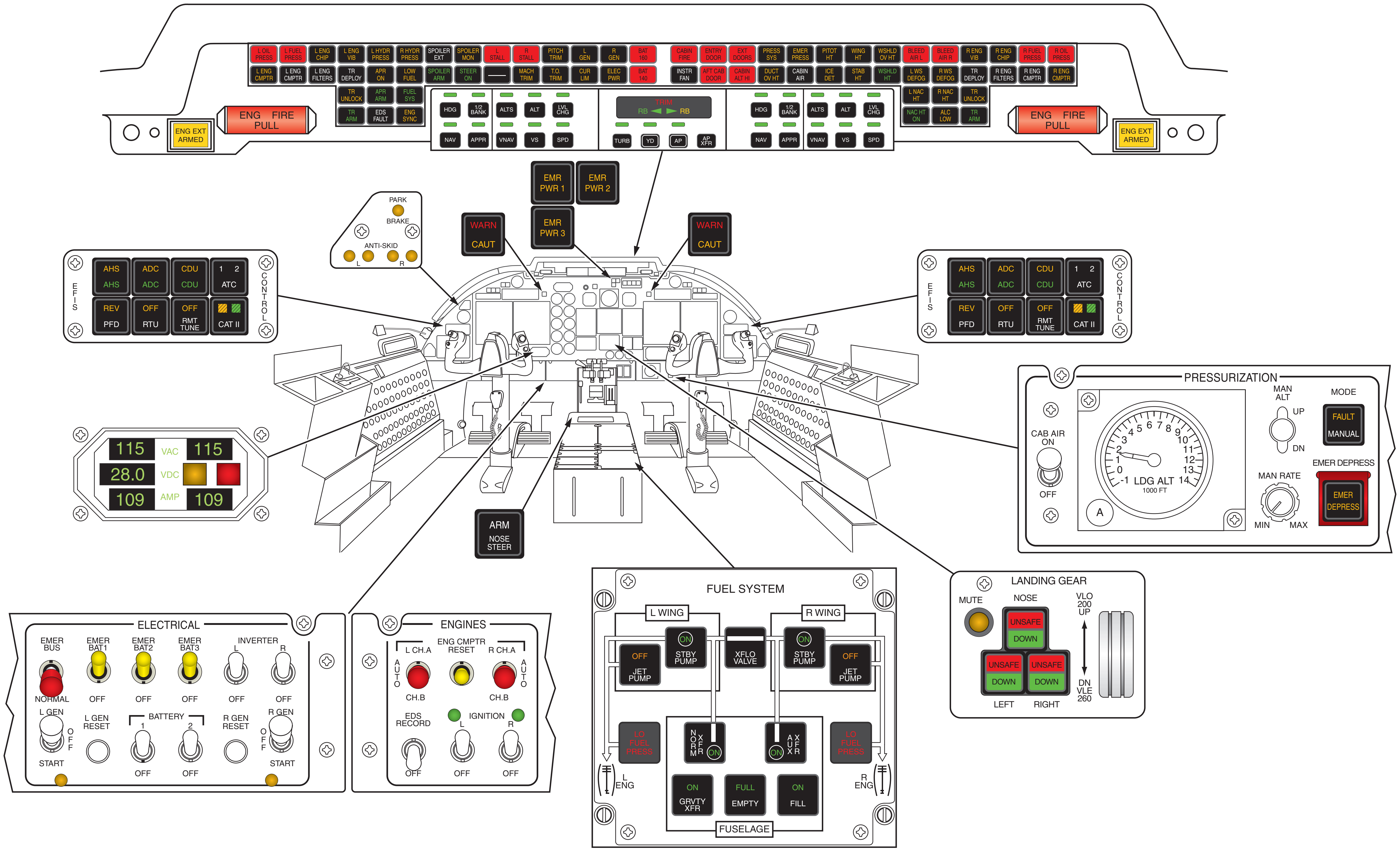


Figure ANN-1. Annunciators